This document is guaranteed to be current only to issue date.

Some Mars Global Surveyor documents that relate to flight operations are under revision to accommodate the recently modified mission plan.

Documents that describe the attributes of the MGS spacecraft are generally up-to-date.

Mars Global Surveyor Project

MISSION PLAN

Final Version, Rev. B (MGS 542-405)

November 1996

Jet Propulsion Laboratory
California Institute of Technology
Pasadena CA, 91109

National Aeronautics and
Space Administration

JPL D-12088

Mars Global Surveyor Project

Mission Plan Document

Final Version, Rev. B (MGS 542-405, November 1996)

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Mars Global Surveyor Project

Mission Plan Contents

1. Overview

The Mission Plan for Mars Global Surveyor (MGS) describes a baseline strategy for successfully achieving the scientific objectives of the mission within the capabilities and constraints of the project systems, subject to the project policies on the use of these systems. This strategy will serve as a starting point for planning detailed event sequences, both before launch and during the mission. The Mission Plan also serves as the basic reference for a detailed description of the mission. It documents the major design features and options considered by the Project along with the rationale for major technical decisions.

1.1 Scope

The Mission Plan Document for Mars Global Surveyor responds to the high-level project policy and requirements documents. Accordingly, the Mission Plan does not originate new mission and system requirements. It describes a strategy for conducting the baseline mission which is consistent with the requirements, capabilities, and constraints defined in other project documentation. The baseline mission is the planned series of mission events that barring failure, normally proceed from launch to the end of the mission.

Section 2	Overview of Spacecraft Design, Science Instruments, and Baseline Mission
Section 3	Description of Launch Phase
Section 4	Description of Cruise Phase
Section 5	Description of Orbit Insertion Phase
Section 6	Description of Mapping Phase

Release Version and Release Comments

This November 1996 Mission Plan release is an update to the Final Version (released September 1995) and supersedes all previous releases. The following list summarizes the major changes between this Mission Plan and the previous release.

a) Mission event calendar updated to be consistent with a 6 November 1996 launch date and new end of mission date on 1 August 2000. ΔV budget also updated accordingly.

b) Mission synopsis, spacecraft description, and payload description completely rewritten to be more clear for those not directly involved with the project's design.

c) Gravity wave search campaign added to the mission schedule.

d) Launch targets and lift-off times updated to be consistent with the final report from McDonnell Douglas.

e) Launch sequence of events and post-separation sequence of events updated to match latest sequence design from Lockheed-Martin.

f) Post AB1 burn (first aerobrake maneuver) periapsis altitude is now 150 km.

g) Aerobrake walk-in profile updated to be consistent with “critical scale height” maneuver strategy.

h) Aerobrake walk-out profile updated to be consistent with two-day orbit lifetime.

i) Orbital elements for mapping updated using the Mars 50c gravity field model.

j) Science campaign schedule updated and now includes new 29-day campaign at the start of the mapping phase. This campaign will primarily be dedicated to production of the MOLA global topography map.

k) Data rate usage profile during mapping updated to reflect latest telecom performance profile numbers.

Disclaimer

The information presented in this document represents the final state of the project’s mission design efforts and is current as of 31 October 1996. However, some design issues may be examined in more detail during the cruise phase of the mission. Any changes will be documented in change requests or in technical memorandums.

1.2 Relationship to Other Documents

The Mission Plan is consistent with and responsive to the requirements and objectives of the following project documents:

Mars Global Surveyor Project Plan	(542-010, December 1994)
Investigation Description and Science Requirements Document	(542-030, February 1995)
Mission Requirements Document	(542-400, September 1995)
Planetary Protection Policy	NASA NHB 80201.12B

Based on the best information available as of 31 July 1996, the Mission Plan is consistent with the capabilities and constraints of the project systems as described in the following documents:

Spacecraft Requirements	(542-200, October 1994)
Mission Operations Specification	(542-409, September 1995)
Navigation Plan	(542-406, September 1995)
Planetary Protection Plan	(542-402, October 1996)

Some more detailed aspects of the mission design for Mars Global Surveyor are described in the following documents:

Trajectory Characteristics Document	(542-410, September 1995)
Delta 2 Target Specification	(542-411, June 1995)
Mars Observer Planetary Constants and Models	(642-321, November 1990)
Mission Sequence Plan	(542-407, September 1995)
Detailed Mission Requirements on the DSN	(542-424, September 1996)

1.3 Update History

During the development phase of the mission, updates for the Mission Plan occurred as specified by the following schedule. Future revisions may be released during flight if deemed necessary by the Flight Operations Manager. However, there are no further releases scheduled at this time.

Draft Version	August 1994
Preliminary	September 1994
Draft for Final	June 1995
Final Version	September 1995
Final, Revision A	July 1996
Final, Revision B	November 1996

1.4 Acknowledgments

The authors of the mission plan are extremely grateful to the following members of the MGS team who provided information and inputs critical to the writing of the plan:

Joe Beerer, Bill Blume, Daren Casey, Dan Johnston, Dan Lyons	Mission Design (JPL)
Nick Smith, Jim Taylor, Wayne Sidney, Bill Willcockson	Mission Design (LMA)
John Callas, Mick Connally, Tom Thorpe	MGS Science Office (JPL)
Gene Bollman, Pat Esposito	Navigation (JPL)
Stan Butman, George Chen, Charles Whetsel	Spacecraft & Subsystems (JPL)

1.5 Questions or Comments?

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2. Mission Background

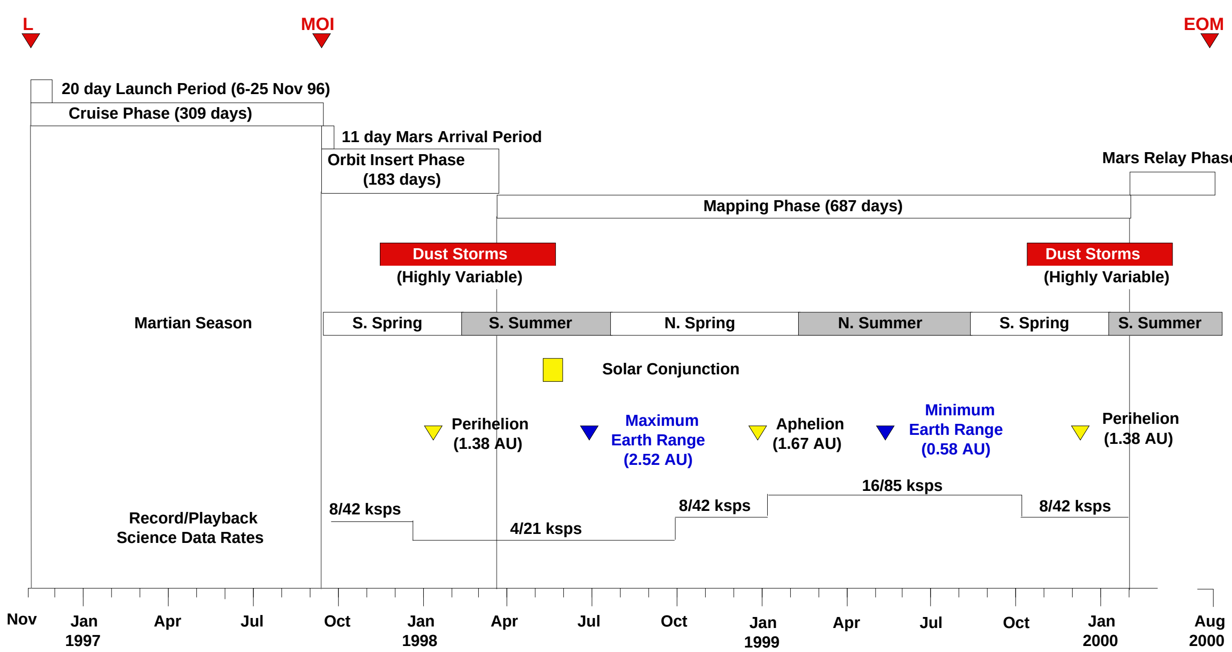
In November 1996, NASA and the Jet Propulsion Laboratory will begin America's return to Mars after a 20-year absence by launching the Mars Global Surveyor (MGS) spacecraft. The MGS mission will recover most of the lost objectives of the 1992 Mars Observer mission by delivering a single spacecraft to the red planet for a two-year study of Mars' surface, atmosphere, and gravitational and magnetic fields. Achieving the scientific objectives of the MGS mission will require placing the spacecraft in a low-altitude, near-polar, Sun-synchronous orbit around Mars and returning data over a complete Martian year. This document will provide an overview of the MGS mission plan, spacecraft, and science instruments.

2.1 Mission Synopsis

A Delta 2 (7925A) launch vehicle will boost the 1,062-kilogram Mars Global Surveyor spacecraft from Cape Canaveral Air Station (CCAS) during the November 1996 launch opportunity (see Figure 2-1 on for the overall mission timeline). The spacecraft will utilize a Type-2 transfer trajectory with a trans-Mars flight time of about ten months. After arriving at the red planet in September 1997, MGS will be propulsively inserted into an initial, highly elliptical capture orbit with a period of 48 hours. Over the next five months, the spacecraft will be gradually lowered into the mapping orbit by the use of aerobraking. This technique works by dipping the spacecraft into Mars' upper atmosphere on every periapsis passage in order to slow down and lower apoapsis.

Mars Global Surveyor will utilize a Sun-synchronous mapping orbit at a 378-km index altitude, and with a descending node orientation of 2:00 p.m. with respect to the fictitious mean Sun. In this 92.9^{o} inclination orbit, the MGS spacecraft will circle the red planet once every 117.65 minutes. Once every seven Martian days (sols), the spacecraft will approximately retrace its ground track. After each seven sol cycle (88 orbits), the ground track pattern will be offset eastward by 59 km from the tracks on the previous cycle. This orbit will provide a repeat cycle scheme that will allow 99.9% global coverage to be built up from repeated instrument swaths across the planet.

Figure 2-1: ***General Timeline for MGS Mission***



Repetitive observations of the planet's surface and atmosphere from the mapping orbit will be conducted over a time span of one complete Martian year (687 Earth days), from March 1998 to January 2000. Data returned from six prime experiments on the spacecraft will provide for a better understanding of the geology, geophysics, and climatology of Mars. Five of those six will utilize scientific instruments mounted to the spacecraft. The sixth investigation will collect data about Mars by analyzing the spacecraft's radio signal after it reaches the Earth.

Throughout the entire mapping period, the spacecraft will remain in an orientation with the scientific instruments nadir pointed. Because MGS lacks a scan platform, any scanning capability will be provided by the instruments. The normal sequence of collecting science data will involve recording continuously for 24 hours, and then playing it back through the Deep Space Network (DSN) during a 10-hour tracking pass once every Earth day. Approximately every third Earth day, an additional tracking pass will be scheduled to return high-rate, realtime data.

From the end of mapping until the end of mission on 1 August 2000, MGS is scheduled to support the Mars Exploration Program by relaying data from various landers and atmospheric vehicles back to the Earth through the spacecraft's Mars Relay antenna.

2.2 Mission Phases

Five mission phases have been defined to simplify the description of different periods of activity. These are the launch, cruise, orbit insertion, mapping, and relay. Several time epochs will be used throughout this document for defining activities and the boundaries of some of the mission phases and subphases. Launch (L) denotes the liftoff time, defined to be the instant of ignition of the Delta 2's first stage. Injection (TMI) represents the burn-out time of the Delta 2's third stage, used for the trans-Mars injection burn that places MGS onto a trajectory bound for the red planet. Mars Orbit Insertion (MOI) designates the time that the spacecraft begins the propulsive maneuver to slow down and enter Martian orbit after the completion of its interplanetary trajectory from Earth. End of Mission (EOM) marks the end of ground operations to control spacecraft activities and collect data. Table 2-1 summarizes the dates of the mission phases and some key mission events. The dates in Table 2-1 are specific to a mission that launches at the opening of the launch period on 6 November 1996. Under the current launch period strategy, MGS can launch as late as 25 November 1996.

Table 2-1: ***Overview of Mission Phases***

Event	Date	Comments
Launch	6-Nov-1996 (early afternoon EST)	Launch period opens on 6-Nov and closes on 25-Nov-96
Inner Cruise Phase	6-Nov-1996 to 6-Jan-1997	Communications through LGA only because solar arrays must be pointed at a fixed angle to the Sun
TCM1	21-Nov-1996 (L+ 15 days)	Trajectory correct for injection errors, remove aim-point biasing introduced for Mars planetary quarantine
Outer Cruise Phase	7-Jan-1997 to 11-Sep-1997	Communications through HGA, phase begins when Earth-MGS-Sun angle falls below 60^o
TCM2	21-Mar-1997 (TCM1+ 120 days)	Correct for execution errors from TCM1
TCM3	20-Apr-1997 (TCM2+ 30 days)	Correct for execution errors from TCM2
TCM4	22-Aug-1997 (MOI- 20 days)	Final adjustment to MOI aim point
Mars Orbit Insertion (MOI)	11-Sep-1997 (about 01:15 UTC)	MOI can vary from 11-Sep-97 to 22-Sep-97 depending on exact launch date
Orbit Insertion Phase	11-Sep-1997 to 14-Mar-1998	Begins at MOI, lasts 5 months to reach mapping orbit using aerobraking and propulsive maneuvers
Mapping Phase	15-Mar-1998 to 31-Jan-2000	Mars mapping operations for one Martian year, about 687 Earth days in duration
Relay Phase	1-Feb-2000 to 1-Aug-2000	About 6 months of support for future Mars missions

2.2.1 Launch

After lift-off, the first stage of the three-stage Delta rocket will boost the spacecraft to an altitude of 115 km. From there, the second stage will take over and achieve a 185-km, circular, parking orbit at about L+ 10 minutes. After parking orbit insertion, the booster and spacecraft will coast for between 24 and 37 minutes (variable with launch date) until reaching a position over the eastern Indian or western Pacific ocean. At that time, stage two will re-start and thrust for nearly two minutes to raise the apogee of the parking orbit. Then, spin rockets will spin-up the third stage and spacecraft to 60 r.p.m., followed by third stage ignition. The Delta's third stage, a STAR 48B solid, will fire for 87 seconds to complete the trans-Mars injection (TMI) burn. After completing the TMI burn, but before third stage jettison, a yo-yo cable device will deploy from the STAR 48B to de-spin the spacecraft. Section 3 contains more detail.

2.2.2 Trans-Mars Cruise

Cruise covers the time of ballistic flight between Earth and Mars. The spacecraft will take between 301 and 309 days to reach the red planet on its Type-2 trajectory depending on the Earth departure date within the 20-day launch period. A launch at the open of the launch period on 6 November 1996 will correspond to a Mars arrival date of 11 September 1997, while a launch at the close of the period on 25 November 1996 will result in an arrival on 22 September 1997. During cruise, a set of four trajectory correction maneuvers (TCMs) will adjust the interplanetary trajectory to ensure that the spacecraft reaches the proper velocity and position targets prior to the Mars orbit insertion (MOI) burn.

Inner Cruise

During the first part of cruise, called inner cruise, initial deployment and checkout of the spacecraft and payload will be accomplished, and navigation tracking data will be taken to determine the flight path for the purpose of planning and executing the first of four planned trajectory correction maneuvers (TCMs). TCM1 is scheduled to occur 15 days after launch (L+ 15 days).

In inner cruise, all spacecraft communications with the Earth will occur though the low-gain antenna (LGA). The reason is primarily due to the spacecraft configuration and solar panel geometry. Because the high gain antenna (HGA) sits on the spacecraft in a stowed, body-fixed orientation during cruise, communicating with the Earth through the HGA will require turning the spacecraft to point the antenna directly at Earth. However, such an orientation would push the incidence angle of sunlight on the panels past acceptable levels for minimum power generation. Therefore, communications through the LGA represents the only feasible option.

Outer Cruise

Outer cruise will begin when the spacecraft switches from use of the low gain to the high gain antenna for communications with the Earth. The exact time when the switch becomes feasible depends on when the angle between the Sun and Earth as seen from the spacecraft (SPE) falls to a level low enough to allow good power while the spacecraft is oriented to point the HGA directly at Earth. This angle starts at about 120^o at the time of launch and falls to less than 60^o by 6 January 1997, assuming a launch on 6 November 1996.

Currently, the transition date to switch to the HGA from the LGA will occur on 6 January 1997 for the purpose of planning command sequences. However, this date will be subject to change during flight as the spacecraft team evaluates the telemetry. In the interest of maintaining the highest possible communications link margin with Earth, switch over will occur as early as possible.

Most of outer cruise will consist of minimal activity as the spacecraft transits to Mars. The vast majority of the events will involve acquiring navigation and tracking data to support the remaining TCMs. During the last 30 days of approach to Mars, the focus will be on final targeting of the spacecraft to the proper aim point, and preparations for orbit insertion. Some science observations during this “Mars approach” time period will also occur.

2.2.3 Mars Orbit Insertion

MOI will slow the spacecraft and allow Mars to capture it into an elliptical orbit. Before the burn, the spacecraft’s velocity relative to Mars will measure approximately five kilometers per second. Near the periapsis of the inbound hyperbolic trajectory, the 659-N main engine will fire for between 20 to 25 minutes to provide a ΔV of about 980 m/s. Burn ignition will occur about 10 minutes before periapsis. During the burn, the spacecraft will utilize a “pitch-over” maneuvering strategy to slew the spacecraft at a constant rate in an attempt to keep the thrust nearly tangent to the trajectory arc. After MOI-burn cut-off 10 minutes after periapsis, the spacecraft will orbit Mars on a highly elliptical orbit with a period of 48 hours and periapsis altitude of about 300 km (periapsis radius of 3,700 km).

2.2.4 Aerobraking

The MGS spacecraft will not carry enough propellant to propulsively reach the required low-altitude, Sun-synchronous mapping orbit due to the relatively low interplanetary injected mass capability of the low-cost Delta booster. Consequently, the spacecraft will rely on aerobraking, an innovative mission-enabling technique, to trim the initial, highly elliptical, capture orbit down to mapping orbit altitudes. During aerobraking, the spacecraft will pass through the upper fringes of the Martian atmosphere on every periapsis pass. Friction from the atmosphere during the drag pass will cause the spacecraft to lose a small amount of momentum and will cause the altitude on the next apoapsis pass to slightly decrease. The rate at which the apoapsis altitude decreases will be determined by the amount of drag generated. Aerobraking deeper in the atmosphere will provide greater drag and reduce the orbit faster, but will generate higher spacecraft temperatures and dynamic pressures.

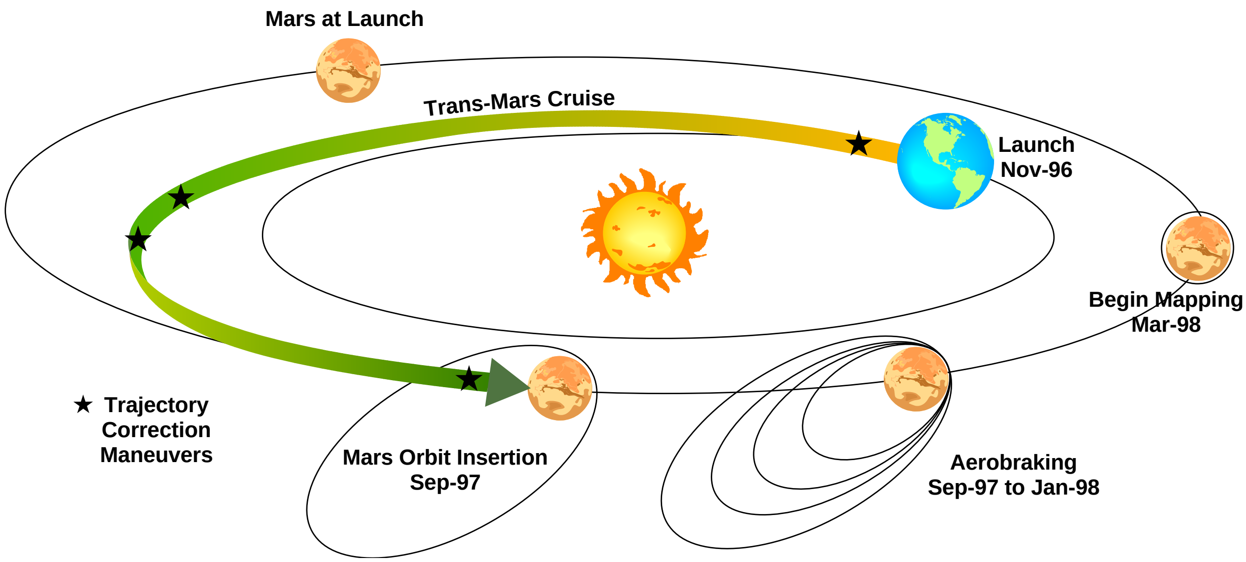
Walk-In

Aerobraking will begin nine days after MOI with the first (AB1) of four to six maneuvers designed to lower the periapsis altitude into the Martian atmosphere in gradual steps. The AB1 burn will be the largest of the six and will lower the periapsis to 150 km. The next manuevers (AB2, AB3, AB4, AB5 and AB6) will provide a further drop to about 112 km. This need for a gradual walk-in is due to both the large uncertainty in the atmospheric density model of Mars, and the time required by the navigation team to gauge the atmospheric density and its orbit-to-orbit variation. Consequently, the navigation team may decide to add or eliminate manuevers as necessary during walk-in operations. All of these burns will be performed using the attitude control thrusters.

Main Phase

After completion of walk-in, the spacecraft will spend about three months in the main phase of aerobraking. During this phase, the apoapsis altitude will shrink in size from about 56,675 km down to 2,000 km. As needed, small propulsive maneuvers (ABMs) executed at apoapsis will maintain periapsis within a well-defined periapsis altitude corridor low enough to produce enough drag to reduce the orbit within the time constraints to reach the 2:00 p.m. node, yet high enough to avoid spacecraft heating and maximum dynamic pressure limits. Due to the oblateness of Mars and the fact that periapsis will be migrating northward toward the pole during main phase, the altitude of periapsis will tend to rise. Consequently, most of the ABMs will be in the down direction to lower the periapsis altitude into the control corridor.

Figure 2-2: ***Schematic Diagram of Key Mission Phases (not to scale)***



Walk-Out

The three weeks of aerobraking following the main phase will represent an extremely critical period as the spacecraft lowers its apoapsis toward the target finish altitude of 450 km. During this time, the spacecraft will slowly "walk-out" of the atmosphere by gradually raising its periapsis altitude to 143 km. Daily ABMs will be performed as necessary to maintain a guaranteed worst-case, two-day orbit lifetime. In other words, in the absence of ABMs due to unforeseen events that inhibit the ability of flight controllers to send commands, the spacecraft will always be at least two days from crashing into the surface.

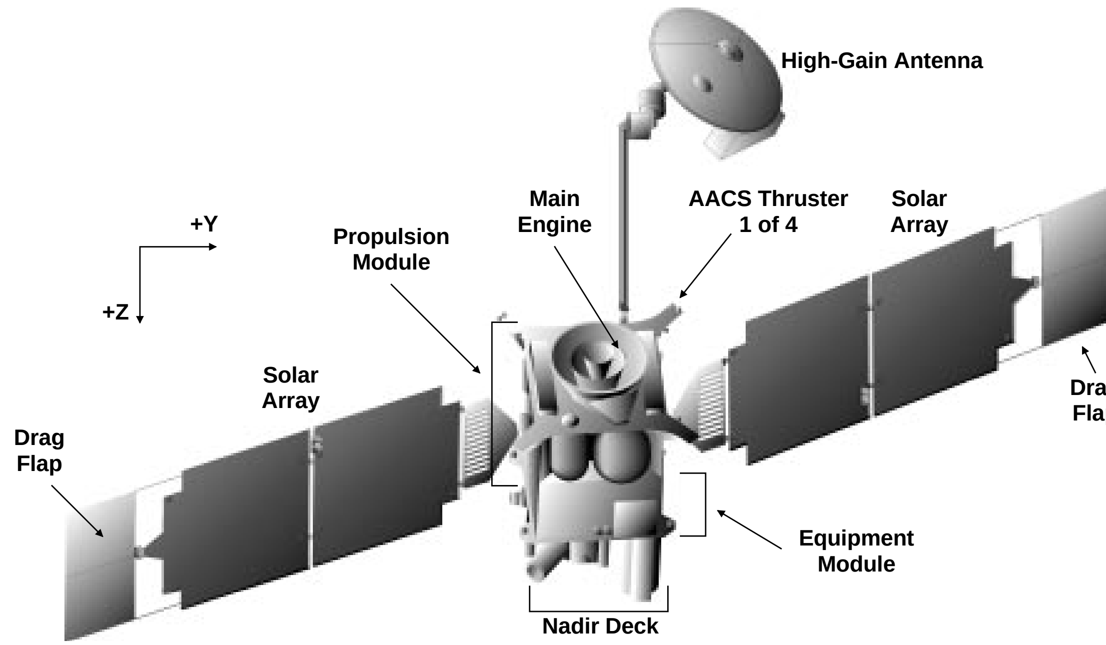
Aerobraking will end with a termination burn (ABX) performed sometime during mid-January 1998. This burn will raise the orbit periapsis out of the atmosphere to an altitude of approximately 450 km. At this time, the spacecraft will be circling in a 400 x 450 km orbit with a period slightly under two hours. In addition, the descending node location will have regressed from its original MOI position at 5:45 p.m. with respect to the fictitious mean Sun to nearly 2:00 p.m.

2.2.5 Transition to Mapping

Transition-to-mapping will begin at the end of aerobraking and lasts until the final mapping orbit has been established and the spacecraft is declared ready to begin mapping operations. After the aerobrake termination burn (ABX), the spacecraft will circle Mars in a transition orbit for a month. During this waiting period, the oblateness of Mars will alter the orbit and cause the location of periapsis to drift to a position almost immediately above the Martian South Pole. At that time, currently scheduled for mid-February 1998, the transition-to-mapping-orbit (TMO) burn will be performed with the intent of "freezing" the periapsis location at the South Pole, and establishing the proper altitude for mapping operations. Throughout this entire transition period between ABX and TMO, the navigation team will conduct gravity calibrations to update the Martian gravity field model using in-flight navigation data returned from the red planet. This update will be crucial toward accurately executing the TMO and other future burns.

An orbit-trim maneuver (OTM) burn will be executed to refine the frozen orbit 12 days after TMO. The 12 days is driven by the navigation team's need for four days to track the spacecraft after the TMO burn and another eight days to plan the maneuver. After the OTM, a ten-day spacecraft deployment and checkout period will follow to allow the operations team time to configure the spacecraft and its instruments for mapping operations.

Figure 2-3: ***View of the MGS Spacecraft in Mapping Configuration***



2.2.6 Mapping

Mapping phase represents the period of concentrated return of science data from the mapping orbit. This phase will start on 15 March 1998 and last until 31 January 2000, a time period of one Martian year (687 Earth days). These dates will remain fixed and are independent of the actual day of lift-off within the launch period. During this phase, the spacecraft will keep its science instruments (+Z panel of the spacecraft) nadir pointed to enable data recording on a continuous basis. On a daily basis, the spacecraft will transmit 24 hours of recorded data back to Earth during a single 10-hour Deep Space Network (DSN) tracking pass. An articulating high gain antenna (HGA) on the spacecraft will allow data recording to proceed while downlink to Earth is in progress.

2.2.7 Relay

The relay operations phase will begin at the end of mapping and continue for about six months. During this phase, the spacecraft will function as a relay satellite for various Mars landers and orbiters in support of the Mars Exploration Program. End of mission will occur on 1 August 2000.

Before relay operations begin, a quarantine orbit raise maneuver will elevate the spacecraft to a near-circular orbit with an average altitude of approximately 400 kilometers. This maneuver is needed to reduce the probability of spacecraft impact with Mars before L+ 20 years and L+ 50 years to the values required for planetary protection. The primary goal of the quarantine maneuver is to raise the orbit. If enough propellant remains after the mapping mission, a slight inclination change will also be performed to maintain the spacecraft's Sun-synchronous orientation. However, no project or program requirements exist with respect to repeat cycle characteristics for the relay orbit.

2.3 Spacecraft Description

Lockheed Martin Astronautics is currently building the Mars Global Surveyor spacecraft at their Denver facility. The design for this three-axis stabilized spacecraft was primarily derived from the Mars Observer spacecraft, with necessary modifications made to support aerobraking, to incorporate the action plans from the Mars Observer failure reports, and to fit onto the Delta 2 launch vehicle. When fully loaded with propellant at the time of launch, the Mars Global Surveyor spacecraft will weigh 1,062.1 kilograms under the current design and ΔV budget. In order to meet this target mass, the spacecraft structure will consist of lightweight composite material divided into four sub-assemblies known as the equipment module, the propulsion module, the solar array support structure, and the high-gain antenna support structure (see Figure 2-3).

2.3.1 Equipment Module

For the most part, the equipment module houses the avionics packages and science instruments. The dimensions of this rectangular shaped module measure 1.221 x 1.221 x 0.762 meters in the X, Y, and Z directions, respectively. With the exception of the Magnetometer, all of the science instruments will be bolted to the nadir equipment deck, mounted above the equipment module on the +Z panel. The Mars Relay antenna is the tallest instrument and extends up 1.115 meters above the nadir equipment deck.

Flight Computers

Inside the equipment module, two redundant flight computers will orchestrate almost all of the spacecraft's flight activities. Although only one of the two units will control the spacecraft at any one time, identical software will run concurrently in the backup unit. Each computer control unit consists of a Marconi 1750A microprocessor, 128 K RAM for storing command sequences as long as six weeks in duration uploaded from Earth, and 20 K PROM that contains code to run basic survival sequences upon entry into fault protection mode due to anomalous conditions.

Additional storage space for science and engineering data will be provided by two solid-state recorders, each with a 1,500 Mbit capacity. The MGS mission will represent America's first interplanetary spacecraft to exclusively use RAM instead of a tape recorder for mass data storage. This technological improvement will dramatically reduce operational complexity, thereby reducing mission planning costs during flight.

Attitude Control

The equipment module also contains three reaction wheels mounted in orthogonal directions to provide spacecraft pointing control authority for all mission events, except for major propulsive maneuvers. A fourth wheel mounted in a direction skewed to the other three will serve as a redundant unit. Attitude information for the spacecraft will be provided by Sun sensors, an inertial measurement unit, Mars horizon sensors, and a star scanner. Table 2-2 describes these devices in more detail.

Table 2-2: ***Spacecraft Attitude Knowledge Devices***

Device	Location	Comments
Sun Sensors	Solar Panels	Provides vector to the Sun, used to begin attitude re-initialization in the event of an anomaly.
Inertial Measurement Unit (IMU)	Inside Equipment Module	Contains gyroscopes and accelerometers that measure angular rates and linear accelerations. Used to determine yaw attitude during Mars mapping, and to aid spacecraft in pointing inertially for both fixed pointing and attitude slews during maneuvers.
Mars Horizon Sensor (MHSA)	Nadir Equipment Deck	Looks at atmospheric horizon during Mars mapping to define the nadir direction by sensing roll and pitch errors.
Star Scanner (CSA)	Nadir Equipment Deck	Complements IMUs by scanning distant stars to provide inertial attitude data.

2.3.2 Propulsion Module

The propulsion module serves as the adapter between the launch vehicle and contains the nitrogen tetroxide (NTO) and hydrazine tanks, main engine, propulsion feed system, and attitude control thrusters. This module bolts beneath the equipment module on the -Z panel and consists of a rectangular shaped box 1.063 meters on each side, with a 0.310 meter tall cylindrical shaped launch vehicle adapter extending from the bottom of the box. Each corner of the box portion of the module contains a small metal protrusion that houses attitude control thrusters. Including the length of these protrusions, the diagonal widths of the propulsion module measure 2.464 and 2.394 meters long.

The main engine, used for large maneuvers such as major trans-Mars trajectory corrections (TCMs) and Mars orbit insertion (MOI), will burn a bi-propellant combination of NTO and hydrazine, and will deliver an I_{sp} of 315 to 318 seconds at a thrust level of 659 N. During main engine burns, four rocket-engine modules (REMs), each containing three 4.45 N thrusters (two aft facing, and one for roll control), will burn hydrazine in mono-propellant, pulse-on mode to provide attitude control. In addition, these mono-propellant thrusters will also be used in pulse-off mode for small trajectory corrections during cruise and orbit trim maneuvers at Mars, and for unloading momentum from the reaction wheels. This propulsion system differs from a conventional bi-propellant system in that the same hydrazine tank will serve both the main engine and attitude-control thrusters, rather than using a separate hydrazine tank for each system.

In total, the propulsion system will provide the spacecraft with a ΔV potential of 1,281.6 m/s. This budget assumes a spacecraft dry mass of 673.73 kg and a total mass of 1,062.1 kg. Table 2-3 shows the ΔV budget for each maneuver in the mission. “Translational” stands for the translational ΔV that affects the trajectory parameters, “Rotational” stands for the rotational ΔV used for attitude control during large maneuvers, and Δi stands for inclination change. All velocities are expressed in meters per second.

Table 2-3: ***Mission ΔV Budget (Launch on 6 November 1996)***

Phase	Maneuver & Type	Translational	Rotational	Comments
Cruise	TCM 1 and 2 (bi) TCM 3 and 4 (mono)	36.0 2.0	0.3	Values based on 95% statistical confidence level
Orbit Insertion	MOI (bi)	968.2	5.9	Pitch-over burn, ΔV includes finite burn loss, Rp = 3,700 km, T = 48 hours
Aerobrake Walk-in	AB1 (bi) AB2 - AB6 (mono)	7.5 2.5	0.1	1.5 m/s reserved for Δi, Hp = 150 km Used to drop periapsis to about 112 km
Aerobrake Main Phase	ABM (mono)	5.0	5.0	Corridor control burns
Aerobrake Walk-out	ABM (mono)	20.0	30.0	Used to slowly raise periapsis
Transition to Mapping	ABX (bi) TMO (bi) OTM1 (bi)	60.2 16.1 7.5	0.4 0.1	Raise periapsis to about 400 km Establish mapping orbit Refine frozen orbit
Mapping	OTM Drag (mono) Momentum Unload	3.9	 40.0	Drag make-up Unload reaction wheels
Quarantine Raise	PQ (mono)	22.8		Performed at end of mapping
Relay	OTM Drag (mono) Momentum Unload	1.0	 10.0	Drag make-up Unload reaction wheels
Contingency	Pre-launch reserve (bi) AB Pop-up (bi/mono)	20.1 14.5 / 2.5		Used for aerobrake and other emergency situations
Total (bi-propellant) **Total (mono-propellant)**		1130.1 59.7	 91.8	Grand total is 1,281.6 m/s for the entire mission

2.3.3 Power

Two solar arrays, each 3.531 meters long by 1.854 meters wide, will provide energy for the MGS spacecraft. Each array mounts close to the top of the propulsion module on the +Y and -Y panels, near the interface between the propulsion and equipment modules. Including the adapter that holds the array to the propulsion module, the tips of the arrays extend 4.270 meters from the sides of the spacecraft. Rectangular shaped, metal drag “flaps” mounted onto the ends of both arrays add another 0.813 meters to the overall array structure. These “flaps” serve no purpose other than to increase the total surface area of the array structure to increase the spacecraft’s ballistic coefficient during aerobraking.

Each array consists of two panels, an inner and outer panel comprised of gallium arsenide and silicon cells, respectively. Available power will start at 1,100 W immediately after launch. During mapping operations at Mars, this amount will vary from a high of roughly 980 W at Mars perihelion to about 660 W at aphelion. When the spacecraft moves into eclipse or turns away from the Sun, energy will flow from two nickel-hydrogen (NiH_2) batteries, each with a capacity of 20 Amp-hours.

2.3.4 Communications

Spacecraft communications with Earth will always utilize X-band frequencies for radiometric tracking, return of science and engineering telemetry, commanding, and radio science experiments. However, the spacecraft’s telecommunications equipment also accommodates Ka-band carrier-only downlink for the purposes of providing a feasibility demonstration. Primary communications to and from the spacecraft will occur through the 1.5-meter diameter high-gain antenna (HGA). From launch until the start of mapping operations at Mars, the HGA will remain body fixed to the spacecraft on the +X side of the spacecraft. Consequently, using the HGA will require slewing the spacecraft to point directly at the Earth. During mapping operations, the HGA will be deployed and will sit at the end of a 2.0-meter boom mounted to the +X panel of the propulsion module. This configuration will allow the HGA to automatically track the Earth by means of two single-axis gimbals that hold the antenna to the boom.

In addition to the HGA, the spacecraft also carries four low-gain antennas (LGA) for emergency communications. Two of the LGAs will function as transmit antennas, while the other two will receive. Placement of these four LGAs will ensure that the spacecraft can receive commands and downlink telemetry over a wide range of attitude orientations. The primary transmit LGA is mounted on the HGA, while the backup is mounted on the +X side of the propulsion module. The two receive LGAs are mounted on the -X panel of the equipment module and the +X side of the propulsion module.

As shown in Table 2-4, the spacecraft’s 25-Watt RF power amplifiers will provide the capability for downlink of science and engineering telemetry at data rates between 21,333 sps to 85,333 sps, depending on the varying Earth to Mars distance. “sps” stands for symbols per second, and a symbol is essentially a Reed-Solomon encoded (250:218 ratio) bit. Therefore, it takes approximately 1.147 bits of storage space to encode one bit of raw data with this encoding ratio.

Table 2-4: ***Data Rate Modes***

Mode	Contents	Realtime Rate	Record Rate	Playback Rate
S&E1	Engineering and Science	4,000 sps	4,000 sps	21,333.33 sps
		8,000 sps	8,000 sps	42,666.67 sps
		16,000 sps	16,000 sps	85,333.33 sps
S&E2	Engineering and Science	40,000 sps	n/a	n/a
		80,000 sps	n/a	n/a
ENG	Engineering Only	2,000 bps	2,000 bps	8,000 bps
		250 bps	n/a	n/a
		10 bps	n/a	n/a

During flight, the spacecraft will generate telemetry using one of three different modes. Depending on the specific data mode (S&E1, S&E2, or ENG), the data can be recorded for later playback, returned in realtime, or both. Table 2-4 lists the different data modes and the data rates that correspond to those modes. As shown in the table, S&E1 is a dual purpose science and engineering data record or realtime mode, while S&E2 is a realtime only mode. Bit allocations for each science instrument will vary depending on the selection of S&E1 or S&E2, and the specific data rate chosen within the mode.

For example, if the S&E1 mode is chosen, data can be recorded and/or downlinked in realtime at rates of 4 ksps, 8 ksps, or 16 ksps. Data recorded for later playback will normally utilize the 21.3 ksps, 42.7 ksps, or 85.3 ksps rates, respectively. This S&E1 strategy corresponds to a 5.333:1 playback to record rate and will allow the spacecraft to return 24 hours of recorded data during a single 10-hour DSN pass.

Data rates for reception of commands from the Earth will vary from as low as 7.8125 bps (emergency situations over the LGA) to as high as 500 bps, with a normal rate of 125 bps. The maximum command reception rate amounts to about 12.5 commands received per second.

2.4 Spacecraft Operating Configurations

Throughout the mission, the Mars Global Surveyor spacecraft will utilize several different operating configurations that depend on the mission phase. Each configuration is characterized by a unique physical state of the spacecraft's appendages (solar array and high gain antenna) and philosophy of choice of attitude to balance power, thermal, and communications constraints. The main modes are launch, array normal spin, maneuver, drag pass, and mapping. Table 2-5 summarizes the configuration modes and associated spacecraft attitude by mission phase.

Table 2-5: ***Configuration Modes by Mission Phase***

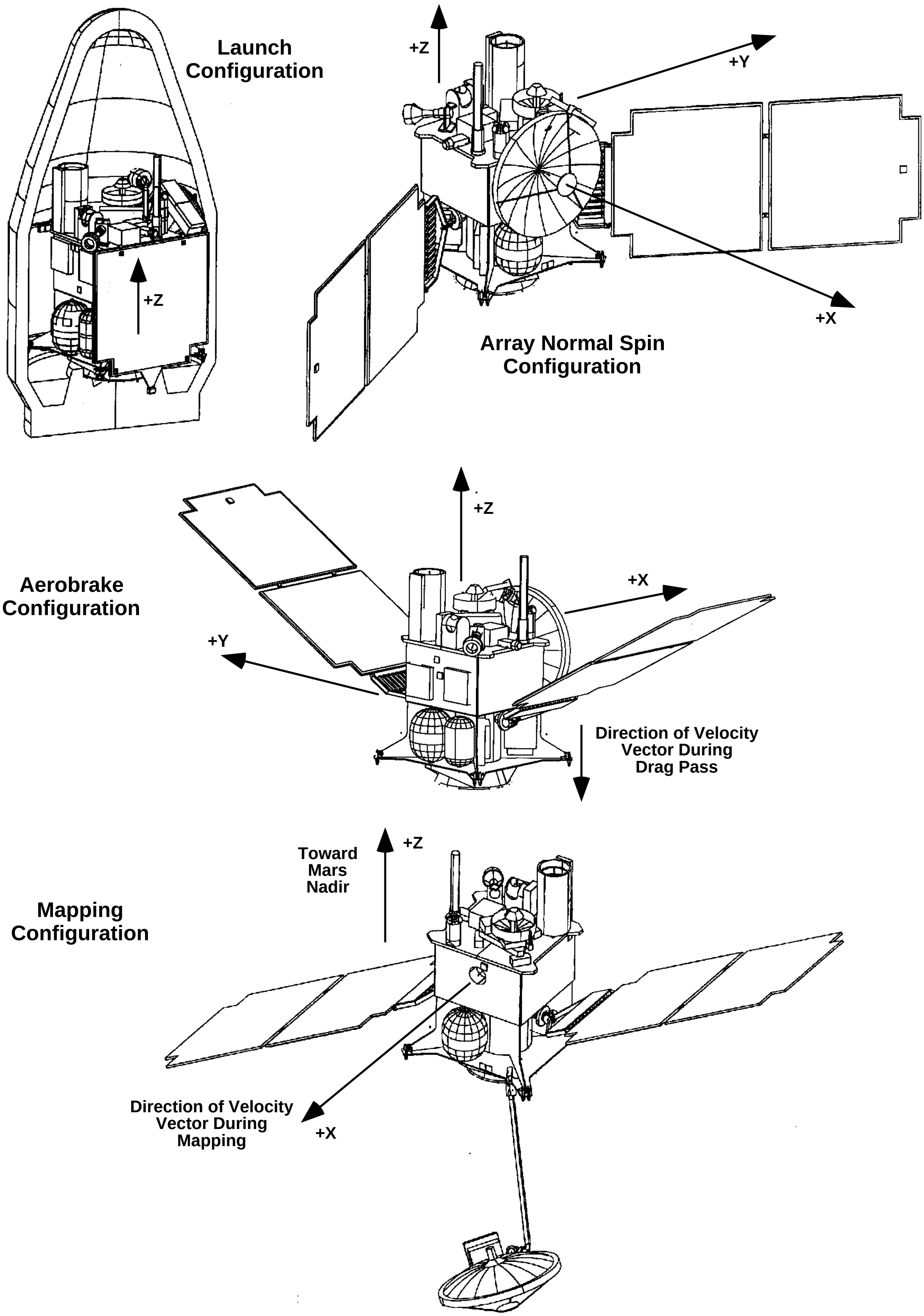
Mission Phase	Normal Configuration Mode	Spacecraft Attitude
Launch	Launch	Z axis pointed along Delta's longitudinal axis, +Z pointed toward top of rocket
Inner Cruise	ANS	+X axis pointed 60^o off Sun, slow roll about +X
Outer Cruise	ANS	+X axis pointed directly at Earth, slow roll about +X
TCMs and MOI	Maneuver	Z axis aligned along inertial direction of thrust vector, +Z in the direction of the desired ΔV.
Orbit Insertion (MOI to mapping)	ANS	+X axis pointed directly at Earth, slow roll about +X
Aerobrake (drag pass only)	Aerobrake	-Z forward along velocity vector, +X nadir pointed
Mapping	Mapping	+X forward along velocity vector, +Z nadir pointed
Relay	Mapping	+X forward along velocity vector, +Z nadir pointed

2.4.1 Launch Configuration

Although the solar panels and HGA attach to the propulsion module, they will be initially stowed and folded upward against the rectangular equipment module at the time of launch. When attached to the Delta rocket, the XY plane of the spacecraft will lie perpendicular to the longitudinal axis of the booster. Since the instruments will be pointed in the spacecraft's +Z direction, they will point upward along the Delta's longitudinal axis, toward the top of the booster's payload fairing during flight (see Figure 2-4).

About half a minute after the Delta jettisons its first stage, it will also jettison the payload fairing and expose the MGS spacecraft to the ambient environment. Because one of the mission flight rules dictates that the +Z axis (science instruments) of the spacecraft can never be pointed within 30^o of the Sun, the Delta must fly an ascent trajectory, parking orbit, and trans-Mars injection profile that keeps the rocket's longitudinal axis at least 30^o from the Sun.

Figure 2-4: ***Spacecraft Configurations (flaps not shown)***



Shortly before the Delta jettisons its second stage, it will spin the third stage and the spacecraft to a rate of 60 revolutions per minute along the booster's longitudinal axis for stabilization purposes. Therefore, the spacecraft will also spin about its +Z axis at the same rate. De-spin will occur several minutes after third stage burn-out. Under normal circumstances, the time during the third stage burn will be the only time that the spacecraft will spin at a rate faster than one revolution every 100 minutes.

2.4.2 Array Normal Spin (ANS)

In array normal spin (ANS), the solar arrays will be swept forward 30^o above the Y axis in the +X direction, and the spacecraft will roll about the +X axis at the rate of one revolution every 100 minutes. This roll serves to maintain a thermal balance on the spacecraft and to constantly slew the star sensor (CSA) across the sky to maintain attitude reference (see Figure 2-4).

The spacecraft's attitude in this configuration will vary by mission phase and will balance power, thermal, and communications constraints. For example, during inner cruise, the +X axis will point to a position halfway between the Earth and Sun. This attitude represents a compromise between needing to point the +X axis directly at the Earth for maximum communications link margin, and needing to point the solar arrays at the Sun for adequate power generation. Communications with the Earth will always occur through the low gain antenna during inner cruise because the undeployed HGA on the +X axis must point directly at the Earth for use.

During all other mission phases that will utilize ANS (outer cruise, capture orbit, aerobrake non-drag pass periods, transition to mapping), the +X axis will point directly at the Earth. Consequently, the high gain antenna (HGA) will be usable because it will also point directly at Earth.

2.4.3 Maneuver Configuration

All maneuvers greater than 10 seconds in duration will use the main engine, located on the bottom (-Z panel) of the propulsion module. During the burns, the solar panels will be swept back 30^o below the Y axis, toward the -Z direction. The reason for this choice is that the thrust direction of the main engine lies along the -Z axis. By sweeping the panels in that direction, as opposed to the +X direction, the spacecraft center of mass will be better aligned with the main engine's thrust axis. In all cases, the active side of the solar panels will also point in the -Z direction. The exact attitude of the spacecraft will be different for each maneuver depending on the location of the thrust vector. For main engine burns, the Z axis must point along the inertial thrust direction.

The spacecraft will use the maneuver configuration for all four trajectory correction maneuvers (TCMs) and Mars orbit insertion (MOI). In addition, this configuration will also be used for all major orbit change maneuvers at Mars such as OTMs, TMO, and ABx burns whether or not the burn is performed with the main engine or attitude control system.

2.4.4 Aerobrake Drag Pass Configuration

The normal spacecraft configuration during aerobrake drag passes will resemble the maneuver configuration, except the solar panels will point in the opposite direction. In this case, the panels will be swept 30^o above the Y axis, toward the +Z direction, and the active side of the solar array will point in the +Z instead of the -Z direction (see Figure 2-4). Because the spacecraft will fly through the Martian atmosphere -Z axis (main engine end) forward during the drag pass, this orientation will maximize protection for the arrays by keeping their active side away from the oncoming airflow. In addition, the +X axis will remain nadir pointed throughout the duration of an aerobrake drag pass.

For all propulsive maneuvers and for the aerobraking drag passes, the spacecraft will turn under three-axis control to the proper attitude. Then, upon completion of the maneuver or drag pass, the spacecraft will return to the configuration required for the current mission phase. Usually, that configuration will be array normal spin.

2.4.5 Mapping Configuration

After insertion into the mapping orbit after the aerobrake phase of the mission, the MGS spacecraft will be configured for mapping operations. In this mode, the spacecraft will be three-axis controlled, using input from the horizon sensors to maintain the science instruments on the +Z panel pointed in the nadir direction. Also, the +X side of the spacecraft will point forward in the direction of orbital motion, and the spacecraft will complete one revolution about the Y axis once per orbit (see Figure 2-4).

Because the mapping orbit is Sun-synchronous with respect to 2:00 p.m. of the fictitious mean Sun, the Sun will always shine (except during eclipse periods) on the +Y spacecraft side at an angle that varies between 50^{o} and 74^{o} from the +Y axis (or 16^{o} to 40^{o} from +X), depending on the Martian time of year. The solar arrays gimbal drive control will be enabled to automatically track the Sun as the spacecraft progresses around the orbit. In addition, the HGA boom will be deployed and its gimbal drive control will allow the antenna to track the Earth around each orbit. This configuration will allow the HGA to point directly at Earth without rotating the entire spacecraft.

2.4.6 Contingency Configurations

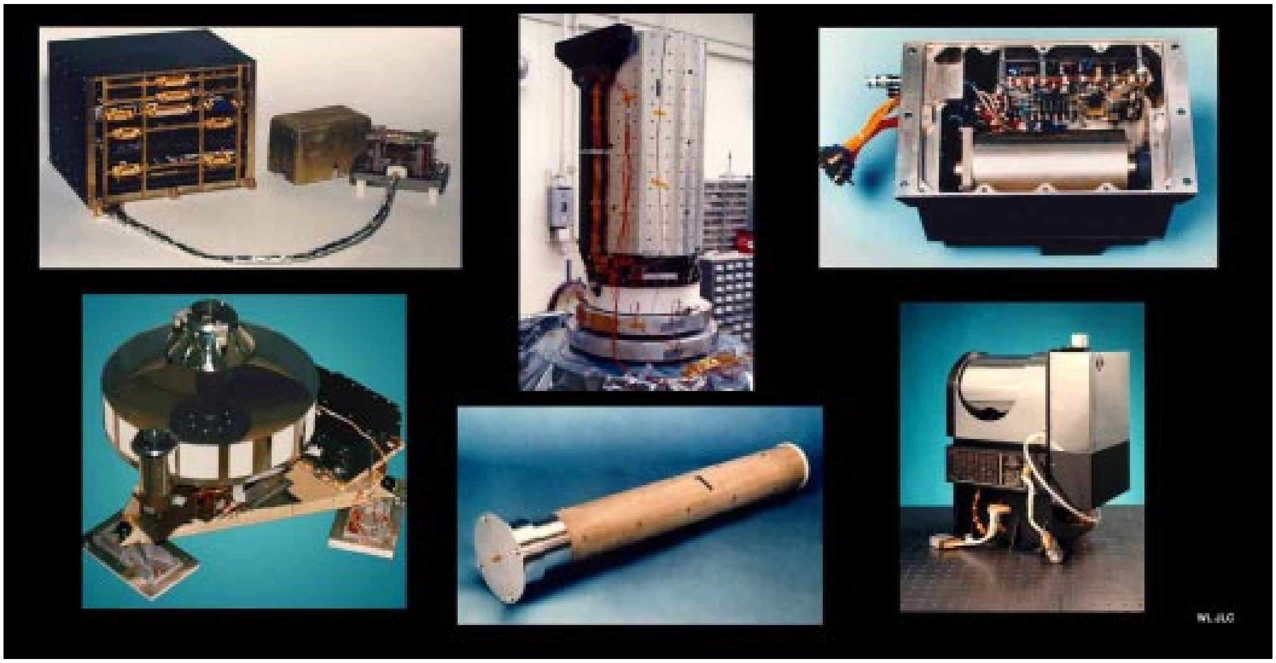
These configurations will allow the spacecraft to regain inertial attitude knowledge in the event that an anomaly causes entry into safe or contingency mode. Upon entering one of these two modes, the spacecraft will automatically transition to a “Sun coning” mode where it will find the Sun, off point the +X axis from the Sun by a pre-selected angle, and then roll about the vector to the Sun at the rate of one revolution every 100 minutes. The key to this mode is that once attitude knowledge has been lost, the only inertial reference that the spacecraft can automatically obtain is the vector to the Sun as determined from the Sun sensors.

After transitioning to Sun coning, the spacecraft will wait until communications has been re-established with the ground. At that time, the control team can command the spacecraft to transition to SUN-STAR-INIT. This mode looks exactly the same as Sun coning except for the fact that the star sensors will scan the sky to reinitialize attitude reference as the spacecraft rolls.

During the cruise, orbit insertion, mapping, and relay phases of the mission, the +X axis offset angle from the Sun for coning will be zero degrees. Consequently, the spacecraft will spin Sun pointed along the +X axis. However, because the high gain antenna (HGA) also points along the +X axis, communications with the ground will occur only through the LGA during Sun coning and SUN-STAR-INIT as using the HGA requires pointing it directly at the Earth.

The only time the spacecraft will take on a different attitude during Sun coning and SUN-STAR-INIT will be immediately after separation from the Delta’s third stage. At that time, the spacecraft will not know its inertial attitude because the third stage will have been spinning at a high rate that will saturate the gyroscopes in the spacecraft’s inertial measurement units (IMUs). During this initial attitude initialization, the spacecraft will point its +X axis 60^{o} off of the Sun. As the spacecraft spins around the Sun line, the +X axis will trace a path around an imaginary cone with a half-angle of 60^{o} and longitudinal axis along the vector to the Sun. In this configuration, the spacecraft +X axis is said to be “coning 60^{o} off the Sun.”

Figure 2-5: ***Science Instruments*** **(CCW from top left: MAG/ER, MOLA, MR, TES, USO, MOC)**



2.5 Science Payload

Mars Global Surveyor will carry six of the original eight scientific instruments flown on the Mars Observer mission. These scientific devices were either Mars Observer flight spares or built from spare components. During 687 days of mapping operations at Mars, the science payload will generate more than 700 Gbits of raw data to support the fulfillment of the five basic goals of the mission: to characterize the surface morphology at high spatial resolution; to determine the global elemental, thermophysical, and mineralogical character of the surface material; to define the global topographical and gravitational field; to establish the nature of the magnetic field; and to monitor the global weather and thermal structure of the atmosphere to evaluate their seasonal impact on the polar caps, atmospheric dust, and clouds. In addition, the data will support mission and scientific planning for future Mars expeditions with special emphasis on the selection of possible future landing sites. See Figure 2-5 for photographs of the science instruments.

Table 2-6: ***Science Instruments Summary***

Acronym	Full Name	Lead Center	Objective
MAG / ER	Magnetometer and Electron Reflectometer	Goddard Space Flight Center (GSFC)	Intrinsic magnetic field and solar wind interaction with Mars
MOC	Mars Orbiter Camera	Malin Space Science Systems (MSSS)	Surface and atmospheric imaging
MOLA	Mars Orbiter Laser Altimeter	Goddard Space Flight Center (GSFC)	Surface topography and gravity field studies
MR	Mars Relay Radio System	Centre Nationale d'Estudies Spatiales (CNES, France)	Support for future Mars missions, both American and international
TES	Thermal Emission Spectrometer	Arizona State University (ASU)	Mineralogy, condensates, dust, thermal properties, and atmospheric measurements
USO (RS)	Ultra Stable Oscillator for Radio Science	Stanford University (team leader)	Gravity field determination and atmospheric refractivity profiles

The names of the scientific instruments for MGS, the location of the principal investigator's home institution, and the measurement objectives for each instrument appear in Table 2-6. More detailed descriptions of each instrument follow in subsequent paragraphs in this section of the Mission Plan. In addition, Figure 2-6 shows the sizes and shapes of the instrument footprints on the surface of Mars

2.5.1 Magnetometer and Electron Reflectometer (MAG/ER)

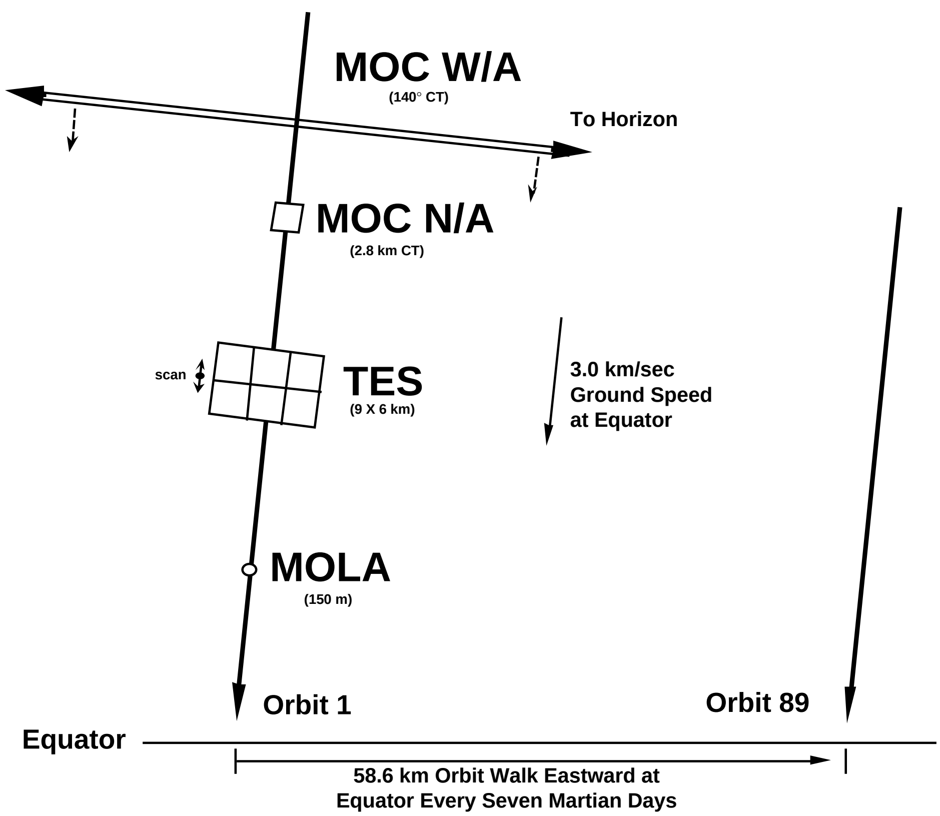
Currently, Mars is the only planet from Mercury to Neptune whose magnetic field has not yet been measured. The design of the solar-array-mounted magnetometers will allow for the in-situ measurement of the global magnetic field in three perpendicular directions over an extremely wide dynamic range between 16 and 65,536 nT. Given this data, the electron reflectometer will determine the local, near-surface magnetic fields by measuring the deflection of ambient electrons (energy range of 1 - 20 keV) by the local magnetic field. Because the paths of these electrons are significantly altered by the magnetic field, use of the electron reflectometer in conjunction with the magnetometer will provide for a resolution capability 10 to 100 times greater than using the magnetometer alone.

Depending on the ground track spacing and the amount of collected data, the Magnetometer will be able to identify surface magnetic features of sufficient strength with a resolution roughly equal to the orbit altitude. With careful calibration, the MGS spacecraft’s contribution to the magnetic field can be subtracted from this data. Both the MAG and ER share a common data processing unit, and the data will be combined and reduced before downlink. Dr. M.H. Acuña of the NASA Goddard Space Flight Center in Greenbelt, MD leads the MAG/ER team.

2.5.2 Mars Orbiter Camera (MOC)

This instrument consists of two independent cameras mounted onto a single assembly. Both cameras are supported by a 32-bit microprocessor for data acquisition and compression, and a 12-Mbyte buffer

Figure 2-6: ***Science Instrument Footprints***



for temporary image storage. The narrow-angle camera employs a 70-cm tall, f/10 Ritchey-Critien reflector with a focal length of 3.5 meters. Inside, two 2048-element charged couple device (CCD) line arrays sit on the focal plane and are mounted in a direction perpendicular to the spacecraft's velocity vector during mapping operations. Two dimensional images of the Martian surface at an unprecedented resolution of 1.4 m/pixel will be formed as the motion of the spacecraft sweeps the detectors forward.

The color-capable, wide-angle camera consists of two f/6, 9.7 mm focal length, 140^o field of view fish-eye lenses feeding into a single focal plane containing two 3456-element CCD line arrays. In order to produce color images, the two lenses use red (575 - 625 nm) and blue (400 - 500 nm) filters, respectively. Wide-angle images of the surface with a resolution of 250 m/pixel at nadir (2 km/pixel at the limb) will be produced in the same "motion swept forward" fashion as the narrow-angle images. These wide-angle pictures will contribute to the MOC's global monitoring mode, an experiment that will provide daily, full-planet observations of the Martian atmosphere and surface similar to the weather pictures of Earth shown during newscasts. Dr. M. Malin from Malin Space Science Systems in San Diego, CA leads the MOC team.

2.5.3 Mars Orbiter Laser Altimeter (MOLA)

The Mars Orbiter Laser Altimeter (MOLA) experiment will generate high-resolution topographic profiles of Mars for studies of geophysical and geological structures and processes. This goal will be accomplished by using a diode-pumped neodymium-yttrium aluminum-garnet laser that will fire 45-mJ pulses of light at the Martian surface at a rate of 10 bursts per second. By recording the time that the pulse takes to reach the surface and bounce back to the instrument's 50-cm Cassegrain collecting mirror, the MOLA team will be able to compute the local altitude under the spacecraft along the ground track.

Each laser spot will measure about 160 meters in diameter on the surface with a spacing of about 300 meters between spots along the ground track. The accuracy in measuring relative topography will vary from one to 10 meters, with an absolute accuracy of about 30 meters. Ultimately, the absolutely accuracy will depend on precise post-reconstruction of the spacecraft orbital position from navigation and radio-science data. Dr. D.E. Smith from the NASA Goddard Space Flight Center leads the MOLA team.

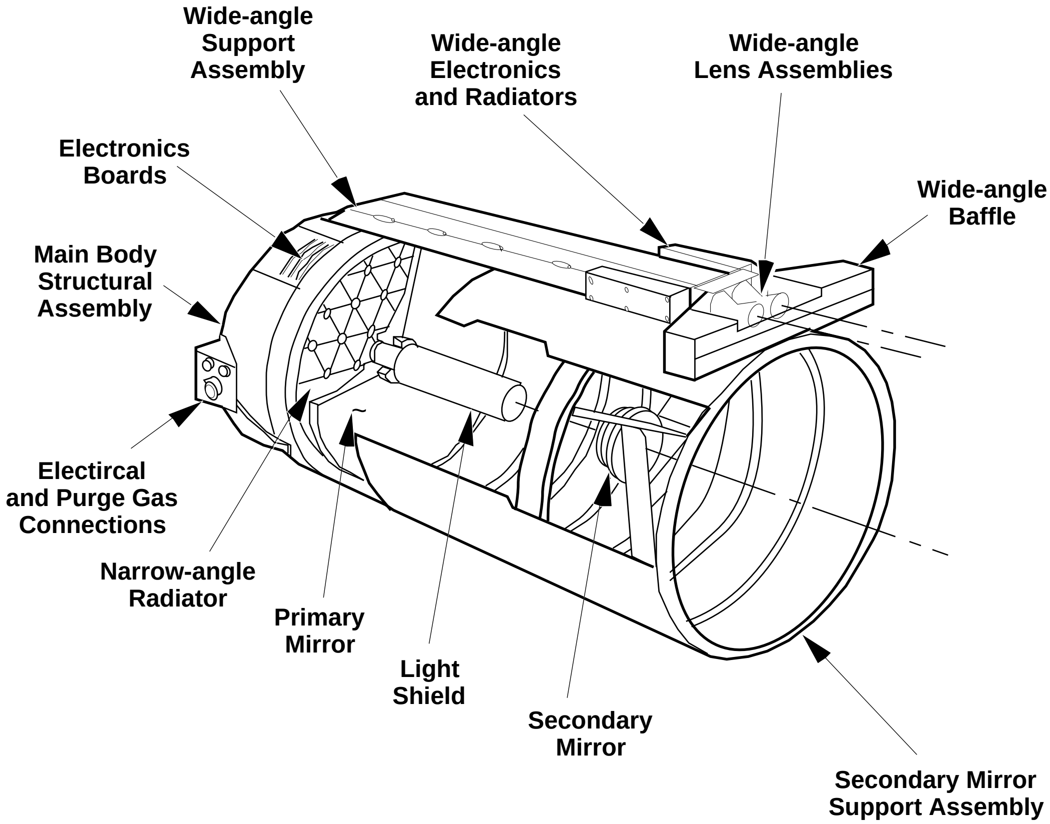
2.5.4 Mars Relay (MR)

The Mars Relay (MR) consists of a radio system and antenna designed to return measurements and imaging data from spacecraft deployed on the surface of the red planet. This instrument consists of a 1-meter tall helix antenna mounted on the nadir panel of the spacecraft and all of the associated electronics. Unlike the main X-band communications system, this device operates at UHF frequencies. The antenna pattern (-3 db) takes the form of a 65^o cone emanating from the tip of the antenna, providing coverage with a 5,000 km effective range for a 8 kbps data rate, and a 1,300 km range for a 128 kbps rate. As the spacecraft orbits Mars, the MR will transmit a 1.3-W, 437.1-MHz beacon to the surface, indicating to the landers that the MGS spacecraft is currently in view. This beacon will serve as an indicator for the landers to begin transmitting their data. The MR is provided by the Centre Nationale d'Estudies Spatiales in France and will be used to relay data from Russian landers (1996 launch) and American landers (1998 launch).

2.5.5 Radio Science (RS)

Radio-science experiments, led by Dr. G.L. Tyler of Stanford University in Palo Alto, California, will advance two fields fundamental to the study of Mars. First, observations of distortions (frequency, phase, and amplitude) in the spacecraft's radio signal as it passes through the Martian atmosphere on the way to Earth will be used to derive high-resolution temperature profiles of the atmosphere with a vertical resolution of 200 meters. Second, by using Doppler tracking to carefully monitor small changes in the frequency of the radio signal from the spacecraft as it orbits Mars, the radio-science team will be able to

Figure 2-7: ***Mars Orbiter Camera (top) and Laser Altimeter (bottom)***



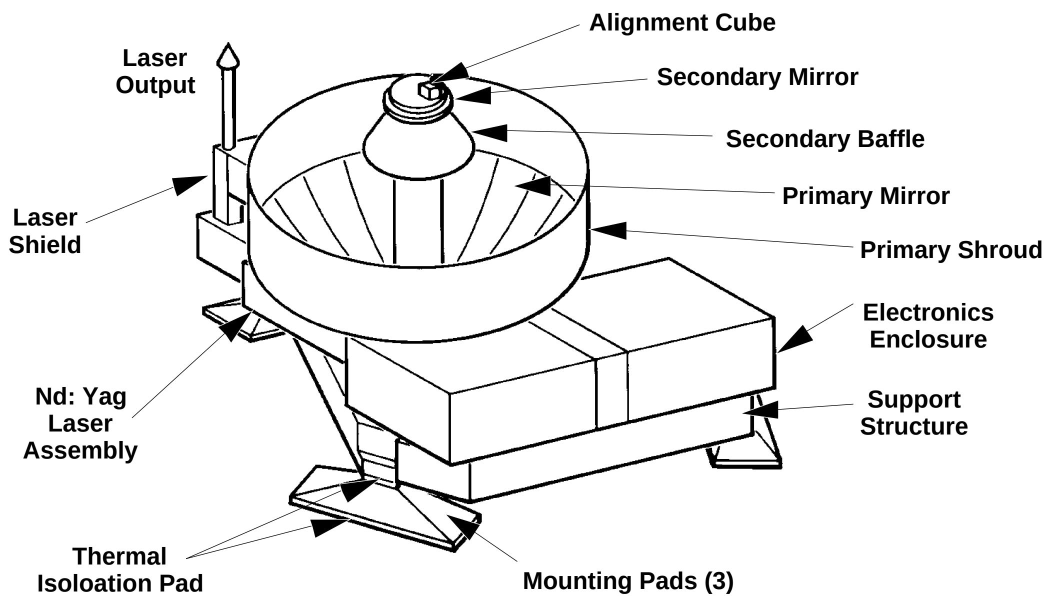
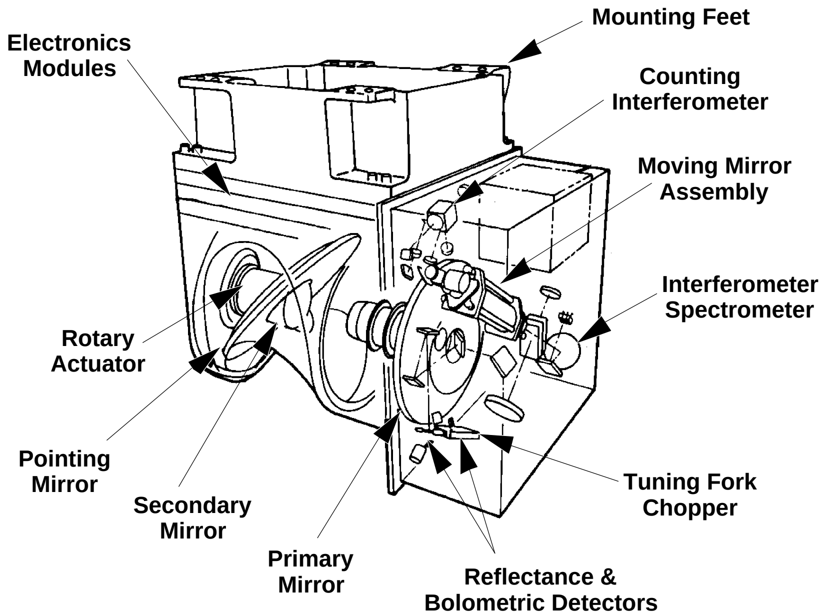


Figure 2-8: ***Thermal Emission Spectrometer (TES)***



reconstruct the Martian gravity field to an unprecedented level of accuracy, perhaps higher than a 50 x 50 field.

Both radio-science experiments will require precise tracking of the spacecraft's radio signal by the antennas of NASA's Deep Space Network. In order to facilitate this requirement, the spacecraft will employ an ultra-stable oscillator (USO) to provide an extremely stable frequency reference for the X-band telecommunications system. This high-quality, low-noise oscillator resonates at 19.143519 MHz and will have a long-term frequency variation limit of less than 1.0 x 10^{-10} Hz.

2.5.6 Thermal Emission Spectrometer (TES)

The Thermal Emission Spectrometer (TES) will function as a combined infrared spectrometer and radiometer designed to measure heat energy radiated from surface and atmosphere of Mars. The investigation team, led by Dr. P.R. Christensen of Arizona State University in Tempe, will use data from TES to determine the thermal and mineralogical properties of the surface, and to learn about Martian atmospheric properties, including cloud type and dust opacity.

This instrument primarily consists of two nadir-pointing telescopes. The larger of the two is a 15.24- cm diameter Cassegrain design that feeds a two-port Michelson interferometer spectrometer with a spectral range from 6.25 to 50 μm. The smaller of the two serves two bolometric channels (0.3 to 3.9 μm and 0.3 to 100 μm) and takes the form of an off-axis parabola shaped telescope. Each telescope utilizes six detectors, each with an 8.3 x 8.3 mrad field of view. Together, the six form a rectangular grid three frames wide (cross-track) and two frames deep (down-track). Although the instrument will normally remain nadir pointed, a rotatable scan mirror will allow the TES' telescopes to view Mars at any arbitrary oblique angle from horizon to horizon.

2.6 DSN Utilization

The 34-meter high-efficiency (34m HEF) antennas of the Deep Space Network (DSN) will provide almost all of the tracking coverage for the mission. This type of antenna was selected for its capability to both transmit and receive X-band signals. Project use of other antenna types, such as the 34-meter beam waveguide (34m BWG), will only be accepted on a negotiated case by case basis.

During periods of normal operation in cruise, mapping, and relay phase, the project's requirement for DSN support are modest at one 10-hour track per day. However, critical operation such as launch, maneuvers (TCMs and MOI), and aerobraking operations will require continuous coverage. In addition, the project will also require continuous coverage for special, week-long science campaigns during mapping to take advantage of special orbit geometry conditions or to observe unique seasonal changes on Mars.

Table 2-7 provides a profile of the tracking support required by the Mars Global Surveyor Project. Stations other than, but equivalent to a 34m HEF, such as a 34m BWG or 70m with comparable up and downlink performance, may only be substituted with negotiation from the project.

Table 2-7: ***DSN Tracking Requirements***

Time Period	Antenna	Tracking Coverage	Data Types
Launch to L+ 30 days (includes TCM1 at L+ 15 days)	34m HEF	Continuous	2-way coherent Doppler and ranging, angular data from launch to L+ 1 day, acquire tracking data as soon as possible after launch.
L+ 30 days to MOI- 90 days	34m HEF	10 hours/pass 1 pass/day	2-way coherent Doppler, 3-way Doppler and ranging.
TCM2 (TCM1+ 120 days) TCM3 (TCM2+ 30 days)	34m HEF	Continuous for a period of 3 days before TCM to 3 days after TCM	2-way coherent Doppler and ranging.
Gravity Wave Campaign (14-Apr-97 to 5-May-97)	34m HEF	Continuous	2-way coherent Doppler and ranging
MOI- 90 days to MOI- 30 days	34m HEF	10 hours/pass 2 pass/day	2-way coherent Doppler, 3-way Doppler and ranging.
MOI- 30 days to start of mapping (includes TCM4 at MOI- 20 days)	34m HEF	Continuous	2-way coherent Doppler, 3-way Doppler and ranging.
MOI- 24 hours to MOI+ 24 hours	70m	Continuous	1-way Doppler and ranging
Routine Mapping Operations (15-Mar-98 to 31-Jan-00)	34m HEF	10 hours/pass 4 pass/3 days	2-way coherent Doppler and ranging, 3-way Doppler and ranging, open loop recording during atmospheric occultations.
Science Campaigns A: 15-Mar-98 to 13-Apr-98 B: 29-Jun-98 to 6-Jul-98 C: 26-Oct-98 to 2-Nov-98 D1: 5-Jan-99 to 12-Jan-99 D2: 20-Jan-99 to 27-Jan-99 D3: 3-Feb-99 to 10-Feb-99 D4: 18-Feb-99 to 25-Feb-99 E: 3-May-99 to 10-May-99 F: 4-Oct-99 to 11-Oct-99 G: 13-Dec-99 to 20-Dec-99	34m HEF	Continuous	Same as during routine mapping
Diametric Occultations Edge-on Orbital Configuration (28-days centered on 28-Oct-98 and 19-Feb-99)	34m HEF	10 hours/pass 2 pass/day (w/ 2 hour overlap)	During the overlap period, simultaneous 2-way coherent Doppler and 3-way Doppler. Otherwise, same as during routine mapping operations.
Communications Relay Phase (1-Feb-00 to 1-Aug-00)	34m HEF	10 hours/pass 1 pass/day	2-way coherent Doppler and ranging

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3. Launch Phase

Launch of the Mars Global Surveyor spacecraft will occur aboard a McDonnell Douglas Delta 2 (7925A) launch vehicle from Space Launch Complex 17A (SLC-17A) at the Cape Canaveral Air Station (CCAS) in Florida. In the past, Delta rockets have primarily been used by NASA and commercial organizations to launch small to medium sized payloads to low Earth or geosynchronous orbits. The current Delta rocket design represents the latest in a long line of highly-reliable and successful family of launch vehicles. This mission will represent the first use of the low-cost Delta to send a spacecraft to another planet, thus producing a savings for NASA of nearly $300 million as compared to Titan launch vehicles used for previous Mars missions.

The launch period for this mission spans a 20-day period from 6 November 1996 through 25 November 1996. Launch phase will start at the beginning of launch countdown and last until separation of the spacecraft from the Delta's third stage following the trans-Mars injection burn.

3.1 Brief Launch Vehicle Description

Delta 7925A launch vehicles consist of five major components that include the three main stages, nine solid rocket motors that attach to the first stage, and the payload fairing. The Delta measures 38.2 meters tall from the tip of the payload fairing to the bottom of the first stage's body. Including the spacecraft (assumed to weigh 1,060 kg for launch vehicle planning purposes), the Delta will weigh approximately 231,325 kg at the time of launch.

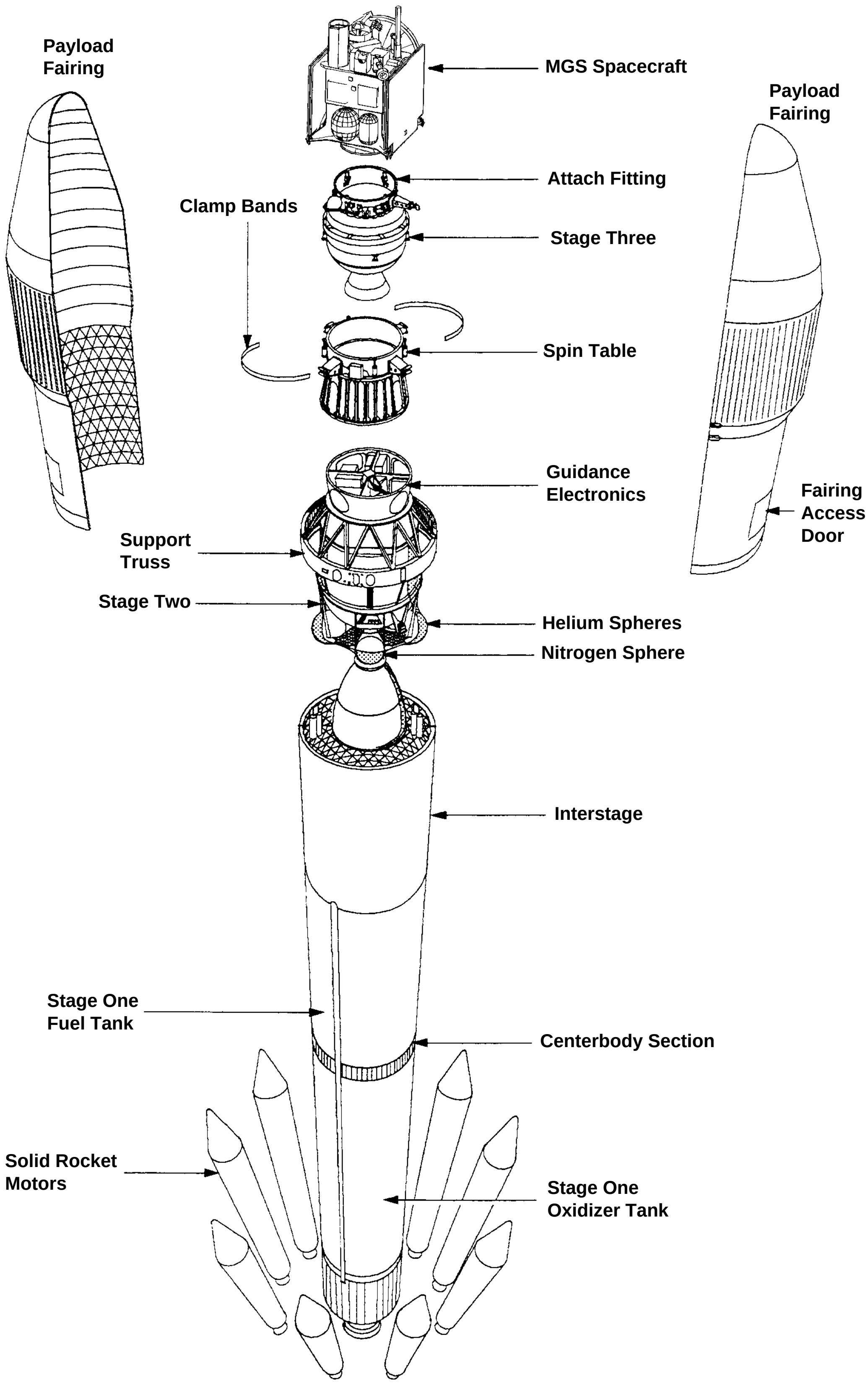
3.1.1 First Stage

Stage one employs a Rocketdyne RS-27A main engine with a 12:1 expansion ratio. This engine is a single start, liquid, bi-propellant rocket that will provide nearly 890,000 N of thrust at the time of lift-off. Its propellant load (95,655 kg) consists of RP-1 fuel (thermally stable kerosene) and liquid oxygen (LOX) for oxidizer. The RP-1 fuel tank and liquid oxygen tank on the first stage are separated by a center body section that houses control electronics, ordnance sequencing equipment, a telemetry system, and a rate gyro. In addition, stage one also employs two Rocketdyne vernier engines. They will provide roll control during the main engine burn, and attitude control between main engine cutoff (MECO) and second stage ignition.

3.1.2 Solid Rocket Motors

A set of nine solid-propellant graphite epoxy motors (GEMs), each fueled with approximately 12,000 kg of hydroxyl-terminated polybutadiene (HTPB) solid propellant, attach to the first stage to provide augmentation thrust. Each GEM will provide an average thrust of 446,000 N. Six of the nine GEMs, the main engine, and the vernier engines will ignite at the time of lift-off, producing a total thrust of about 2,850,000 N. The remaining three GEMs will ignite 65 seconds into flight, shortly after the initial six burnout.

Figure 3-1: ***Parts of the Delta Launch Vehicle***



3.1.3 Second Stage

The Delta second stage uses a re-startable, liquid, bi-propellant Aerojet AJ10-118K engine that consumes a combination of Aerozine-50 fuel (a 50/50 mix of hydrazine and un-symmetric dimethyl hydrazine) and nitrogen tetroxide (N_2O_4) oxidizer. Since this propellant combination is hypergolic, no catalyst or igniter in the engine thrust chamber is required. In total, the second stage will burn nearly 6,000 kg of propellant at an average thrust of 43,370 N. A set of hydraulically activated engine gimbals will provide pitch and yaw control during powered flight, and a nitrogen cold gas jet system will provide the roll authority. In addition, the nitrogen jets will also provide attitude control for the coast phases.

This stage will provide the thrust needed to boost the launch vehicle and spacecraft into low Earth orbit after first stage jettison. Also, the second stage will impart some of the velocity needed to send the spacecraft to an Earth-escape trajectory.

3.1.4 Third Stage and Payload Fairing

A spin-stabilized third stage will provide most of the velocity required to boost the MGS spacecraft from low-Earth orbit to a trans-Mars trajectory. This stage consists primarily of a Thiokol Star-48B solid motor. The engine on the third stage will provide an average thrust of 66,370 N and will burn about 2,000 kilograms of ammonium perchlorate propellant.

During launch and ascent through the lower atmosphere, a 2.9 meter diameter payload fairing will protect the spacecraft and Delta third stage from aerodynamic forces. The fairing will be jettisoned from the launch vehicle at an altitude of approximately 129 kilometers (L+ 286 seconds), shortly after second stage ignition.

3.2 Pre-Launch Activity Overview

Delivery of the spacecraft to Cape Canaveral is currently scheduled for 15 August 1996, after completion of system testing Lockheed-Martin's Waterton facility near Denver, Colorado. The spacecraft will be flown to Florida on a C-17 cargo jet. After arrival, initial inspection and checkout will take place at the KSC Payload Hazardous Servicing Facility (PHSF) where a DSN end-to-end compatibility test via the MIL-71 complex will be conducted. Technicians at KSC will then load the spacecraft with propellants and mate it to the Delta third stage. Following mating at the PHSF, initial interface verification testing will be performed.

Stacking of the Delta first and second stage will occur at Pad-17A in parallel with the activities at the PHSF. Roughly three weeks before launch, the coupled spacecraft and third stage will be moved to Pad-17A for mating with the bottom two stages of the Delta booster. Following final interface testing and close-out activities at the launch pad, the payload fairing will be installed, and Delta launch readiness will be verified.

3.3 Launch Strategy

Designing a launch strategy for Mars Global Surveyor provided many challenges due to the presence of many aspects not commonly found on interplanetary launches. Some of these include near-instantaneous launch windows and fixed azimuth flight profiles during boost. These constraints were dictated by the operational characteristics of the Delta combined with a “heavy” spacecraft relative to the performance capability of the launch vehicle. Nevertheless, the launch strategy developed for this mission will provide a probability of lift-off within the 20-day launch period of greater than 97%.

3.3.1 Launch Window

Delta rockets do not possess the capability to automatically "re-target" their launch azimuth in realtime at the launch pad, nor does the first stage possess the capability to yaw-steer during ascent. Once the guidance software has been loaded, the booster must fly at a single, pre-determined azimuth. Switching azimuths will require manually loading new guidance targets. Consequently, traditional launch windows (typically two hours in duration for the Mars Observer launch) are not possible. Instead, the booster must launch within a very small time period (about ±1 second) when the launch site rotates precisely into the plane containing the parking orbit and the departure asymptote. These extremely short windows are referred to as "instantaneous launch windows" or "instantaneous launch opportunities."

Despite the implementation of the instantaneous launch window, the need to avoid potential orbiting debris in the Delta's flight path may require the booster to launch up to 30 seconds before the calculated lift-off time. If this situation occurs, the Delta will fly exactly the same boost profile as if a normal launch had occurred. The error incurred in the spacecraft's Earth departure trajectory will be removed at TCM1 with a penalty of approximately 5 m/s.

3.3.2 Launch Period

The current mission baseline calls for a launch period that opens on 6 November 1996 and closes 20 days later on 25 November 1996. This time period is bounded on both ends by trans-Mars trajectory C_3 values too high for the Delta to achieve, given the mass of the spacecraft. Over the duration of the launch period, the declination of the departure asymptote (DLA) will vary from a minimum of 20.8^o at the open, and 36.4^o at the close. Geometrical constraints of the interplanetary injection problem dictate that the parking orbit prior to trans-Mars injection must lie at an inclination greater than or equal to the DLA. In order to satisfy this constraint, minimize the number of parking orbits in the launch vehicle targeting specification, and maximize the probability of launch, the Delta will fly at one of three different launch azimuths.

For the first half of the launch period (6 November to 15 November 1996), the Delta will launch at an azimuth of either 93^o or 99.89^o and fly to a low-Earth parking orbit of 28.47^o and 29.82^o, respectively. During the second half of the launch period (16 November to 25 November 1996), the declination of the hyperbolic departure asymptote will exceed the latitude of Cape Canaveral Air Station, and the Delta will launch at a 110^o azimuth (with a dog-leg ascent) to reach a 36.5^o parking orbit to compensate. Although a booster performance penalty normally exists for reaching higher inclination orbits, they will be in part offset by the lower C_3 requirements (minimum of 8.89 km^2/s^2 on 18 November) during the second half of the launch period.

Utilization of two launch azimuths during the first half of the launch period will give the Delta two discrete, instantaneous launch opportunities each day. If the booster misses the first due to minor hardware problems or short term weather violations, the launch team can reload the guidance targets and make a second attempt at lift-off on the same day. This scheme will increase the overall probability of lift-off within the launch period and will specifically increase the probability of a lift-off near the open of the period.

The choice of the two specific launch azimuths for the first half of the launch period was driven by constraints imposed by Delta launch operations. Specifically, the two lift-off time solutions corresponding to the two different azimuths must be separated by no less than 64 minutes and no more than 73 minutes. The former represents the minimum time required for the ground team to reload the guidance targets, and the latter corresponds to the maximum time that the liquid oxygen can remain the first stage tanks without freezing vital parts of the Delta's propellant feed system.

The choice of the launch azimuth for the second half of the launch period resulted from range safety constraints. Normally, rockets launched from the Cape must fly at an azimuth several more degrees south of east than the chosen 110^o in order to reach an inclination of 36.5^o directly. Instead, the Delta first and second stages will each perform a “dog-leg” maneuver to reach the proper parking orbit inclination to avoid violating the range safety constraints.

3.3.3 Launch Opportunity

For any given launch azimuth, two opportunities exist every day for a rocket to launch and inject its payload onto the proper Earth escape trajectory. The primary difference between the two, called the long and short coasts, is the length of time that the Delta must wait in its low-Earth parking orbit before reaching the proper location to perform the trans-Mars injection burn. On any given day (relative to the time when the launch site on Earth rotates through the departure asymptote), the long coast launch opportunity always occurs first.

During the wait in low Earth orbit for the trans-Mars injection burn, the spacecraft will rely on its batteries for power because the solar panels will have not yet been deployed. Preliminary analysis shows that on the long coast, the spacecraft will need to rely on battery power for up to 91 minutes. Although this length of time is undesirably long, battery depth of discharge does not represent the limiting factor in the choice between the two launch opportunities.

The major constraint involves Sun avoidance. An MGS flight rule specifies that science instruments (located on the +Z axis of the spacecraft) must always remain pointed at least 30^o away from the Sun under normal (non-safe-mode) conditions. On this mission, the long coast requires a dawn launch from the Cape. Because launches occur generally in the eastward direction, the science instruments will be pointed almost directly at the Sun at the time of payload fairing jettison. For this reason, the MGS mission will utilize the short coast launch opportunity for all three launch azimuths.

3.3.4 Probability of Commanded Shutdown (PCS)

Probability of commanded shut-down (PCS) for the Delta second stage plays a key role in determining the duration of the MGS launch period. The reason is that the second burn of the second stage provides part of the energy required to place the spacecraft on the trans-Mars trajectory. PCS defines the probability that the second stage engine will complete its burn before the propellant supply depletes. By accepting a lower PCS value, it is possible to achieve a higher injected mass. Current project policy dictates a minimum PCS value of 95% in order to achieve the necessary C_3 for a given launch day. However, under the current MGS launch design, the PCS does not fall below 97% until the end of the launch period.

Predicting the net effect of a PCS violation during flight is difficult. If the propellant supply of the second stage runs out during the stage’s second burn, the third stage will still ignite at the proper time. In this scenario, the total ΔV imparted by the combination of second burn of the second stage and the third-stage burn will fall short of the required amount to achieve the proper C_3 target. If propellant depletion occurs very late during the second burn of the second stage, it is conceivable that the velocity deficit could be corrected at the first trajectory correction maneuver (TCM) or with an contingency TCM. However, such a scheme would result in a significant reduction in total mission ΔV capability and would prohibit flying the mission according to the current baseline plan.

3.4 Boost Profile and Injection

In general, the exact mission elapsed times for key events depend on the orientation and location of the Earth departure asymptote (variable with each launch day), and the launch azimuth of the booster (either 93^o, 99.89^o or 110^o). However, in all cases, the mission elapsed event times for the first stage boost profile will always remain constant. Table 3-1 lists the times for critical events in the boost sequence.

Table 3-1: *Major Events During Ascent*

Event	Time (93^o Azimuth)	Time (99.89^o Azimuth)	Time (110^o Azimuth)
Lift-Off	0.000 seconds	0.000 seconds	0.000 seconds
Mach 1	32.248	32.248	32.248
Maximum Dynamic Pressure	49.410	49.410	49.410
Solid Motor Burn-Out (6 of 9)	63.120	63.120	63.120
Solid Motor Ignition (3 of 9)	65.500	65.500	65.500
Solid Motor Jettison (6 of 9)	66.500	66.500	66.500
Solid Motor Burn-Out (3 of 9)	128.820	128.820	128.820
Solid Motor Jettison (3 of 9)	131.500	131.500	131.500
Stage 1 Main Engine Cut-Off	260.664	260.664	260.664
Stage 1 Jettison	268.664	268.664	268.664
Stage 2 Ignition	274.164	274.164	274.164
Payload Fairing Jettison	286.000	286.000	286.000
Stage 2 First Cut-Off	576.592	577.324	583.573
Stage 2 Restart	2420 / 2812	2213 / 2557	2027 / 2550
Stage 2 Second Cut-off	2548 / 2932	2340 / 2677	2147 / 2672
Stage 2 Jettison	2601 / 2985	2394 / 2731	2200 / 2725
Stage 3 Ignition	2638 / 3022	2431 / 2768	2238 / 2762
Stage 3 Burn-Out	2725 / 3110	2518 / 2855	2325 / 2849
Yo-yo Deploy and Despin	3003 / 3387	2796 / 3132	2602 / 3127
Stage 3 Jettison	3008 / 3392	2801 / 3137	2607 / 3132

In Table 3-1, the appearance of two time values for a single event indicates that the event time varies with the launch date. For the 93^o and 99.89^o flight azimuths, the first and second times correspond to the 6 November and 15 November launch dates, respectively. For the 110^o azimuth, the first and second times correspond to the 16 November and 25 November launch dates, respectively.

During the launch phase, the booster will not provide the spacecraft with any power or telemetry capabilities. The spacecraft will launch with power for the computer, receiver, and attitude control sensors supplied from the batteries. Switch-over from launch-pad power to internal spacecraft power will occur at T- 4 minutes prior to launch.

3.4.1 Stage One

Lift-off will occur from SLC-17A at Cape Canaveral Air Station. This time will vary from as late as 13:37 to as early as 9:30 EST depending on launch azimuth and launch date. At the time of lift-off, the main engines and six of the nine solid rocket motors will ignite. Approximately 63 seconds into flight, the solids will burn-out and be jettisoned. Then, the remaining three solids will ignite and burn for another 63 seconds before being ejected. Main engine cut-off (MECO) will occur 261 seconds after lift-off at a sub-orbital altitude of 115.2 km, 541.3 km down range from the launch site. The vernier engines will continue to burn for another six seconds, and first stage jettison will occur two seconds later at L+ 269 seconds.

3.4.2 Stage Two

At L+ 274 seconds, roughly five seconds after stage one jettison, stage two will ignite. Twelve seconds later, the free molecular heating rate on the vehicle will have dropped to below 1135.0 W/m2, allowing the Delta to jettison its payload fairing and expose the spacecraft to the vacuum of space. In total, stage two will thrust for approximately five minutes to boost the spacecraft from its sub-orbital state at MECO to a circular, low-Earth parking orbit at an altitude of 185 km. Second stage cut-off (SECO1) will occur between L+ 577 seconds to L+ 584 seconds, depending on the exact date of launch and choice of flight azimuth.

3.4.3 Trans-Mars Injection

After parking orbit insertion, the booster and spacecraft will coast for between 24 to 37 minutes (variable with launch date) until they reach the proper position to begin the two-burn trans-Mars injection sequence. First, stage two will re-start and thrust for roughly 120 seconds to raise the apogee of the parking orbit. After cut-off of the second stage engine (SECO2), the Delta will coast for 53 seconds before jettisoning the second stage. During that time, small rockets on the spin table (attached to the bottom of stage three) will fire to spin stage three and the spacecraft to a rate of 60 r.p.m. for spin-stabilization purposes.

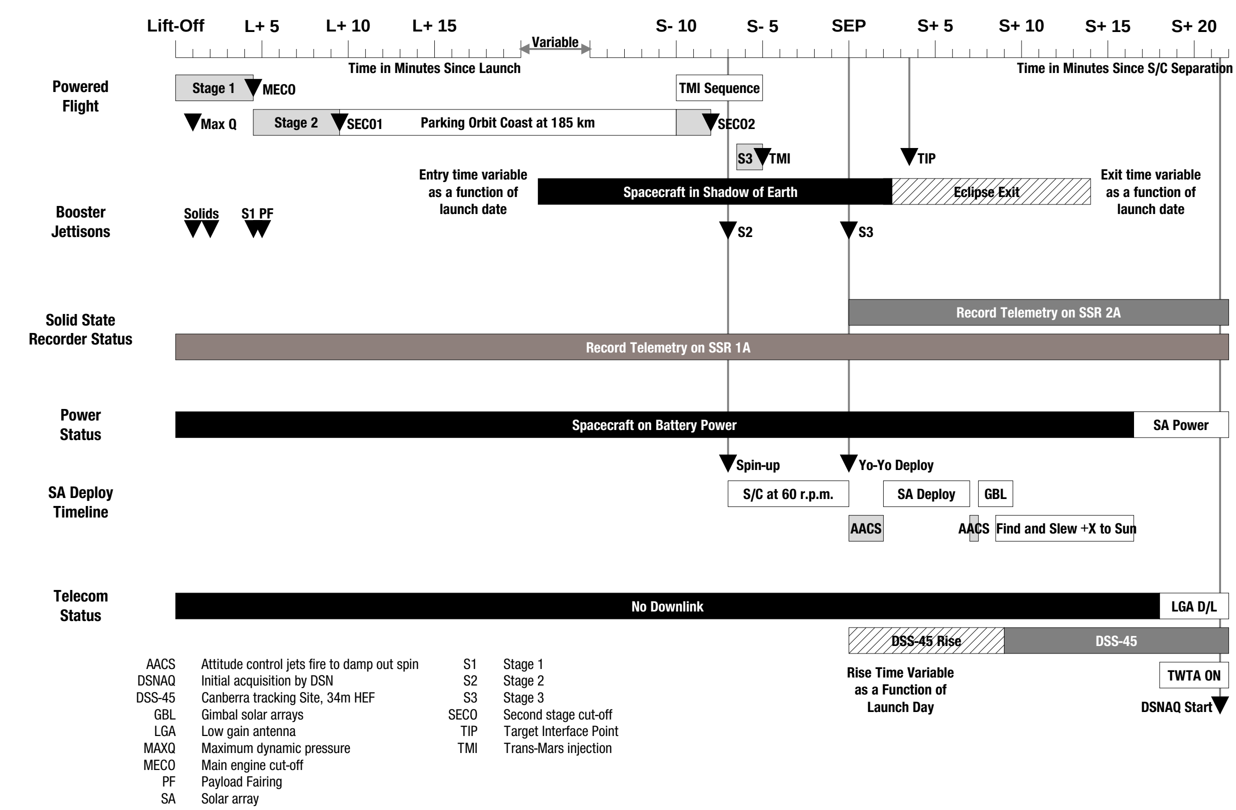
Once 37 seconds have elapsed after stage two jettison (90 seconds after SECO2), stage three will ignite and burn for 87.3 seconds to complete the trans-Mars injection sequence. At the end of the burn, the spacecraft will be on an Earth escape trajectory. For the baseline, short-coast option, trans-Mars injection will almost always occur in darkness, somewhere over the Indian Ocean.

The combination of the second burn of the second stage and the third-stage burn will provide the ΔV needed for trans-Mars injection. During every day of the launch period, the third stage will impart the same amount of ΔV to the spacecraft. The burn time and ΔV of the second stage's second burn will vary depending on the specific C_3 requirements of the given launch day.

3.4.4 Despin and Spacecraft Separation

At the time of stage three burn-out, the Delta and MGS spacecraft will still be spinning at 60 r.p.m. This rotation rate must be nullified because the spacecraft functions on three-axis stabilization. In order to

Figure 3-2: ***Timeline of Launch Events***



despin the spacecraft, a yo-yo cable device on the third stage will deploy approximately 278 seconds after burn-out (365 seconds after ignition). The device works by transferring the angular momentum of the third stage and spacecraft to the cable in a fashion similar to how a spinning figure skater slows her spin by extending her arms.

Five seconds after yo-yo deploy, pyrotechnic devices will fire to sever the connection between the spacecraft and third stage. A set of four springs will then uncoil to impart a relative separation velocity of between 0.6 and 2.4 m/s between the third stage and spacecraft. The 278 second wait after burn-out for separation is designed to allow adequate time for residual thrust from the third stage to tail-off and to ensure that the stage will not collide with the spacecraft after separation. During this waiting period, a set of thermal blankets located on the third stage will protect the spacecraft from thermal soakback.

The actual mission elapsed time of separation depends on the length of time that the spacecraft spends in the low-Earth park orbit before trans-Mars injection and will vary with each launch day. However, separation will always occur 370 seconds after third stage ignition. The choice of 370 seconds is in part driven by the standard cascaded event timers that McDonnell Douglas installs on the third stage.

3.5 Launch Targets

Launch vehicle targets represent the state that the Delta must deliver the spacecraft to in order to place the spacecraft on the proper trans-Mars trajectory. The targets are defined as osculating C_3, DLA, and RLA achieved at the target interface point, defined as 10 minutes after third stage ignition. Table 3-2 lists the launch targets for each launch date. In the table, two values for each targeting parameter are listed for launch dates between 6 November 1996 and 15 November 1996, inclusive. The top number corresponds to the 93.0^o launch azimuth, while the bottom one corresponds to the 99.89^o launch azimuth. For the rest of the launch dates, the Delta will fly at an 110^o azimuth. All of the departure states listed in the table have been biased to satisfy Mars quarantine requirements.

The following definitions and assumptions apply to the launch vehicle departure targets listed in Table 3-2:

C_3 Departure energy or hyperbolic excess velocity squared (km^2/s^2)

LO Lift-off time (hh:mm:ss UTC)

TOF Time from lift-off to target interface point (hh:mm:ss)

DLA Declination of the departure asymptote vector (degrees, EME2000)

RLA Right ascension of the departure asymptote vector (degrees, EME2000)

Table 3-2: *Departure Launch Targets*

Launch Date	Lift-Off Time	Time of Flight	Departure C_3	Departure DLA	Departure RLA
6-Nov-96	17:11:17	0:53:58	10.1656	21.2815	173.2868
	18:15:44	0:50:31	10.1551	21.3062	173.2849
7-Nov-96	17:00:50	0:54:18	9.9846	21.8151	173.3121
	18:05:56	0:50:49	9.9747	21.8430	173.3104
8-Nov-96	16:48:11	0:54:43	9.7984	22.4474	173.1808
	17:54:09	0:51:11	9.7890	22.4782	173.1780
9-Nov-96	16:35:20	0:55:11	9.6412	23.0918	173.1528
	17:42:19	0:51:36	9.6324	23.1246	173.1488
10-Nov-96	16:20:05	0:55:44	9.4837	23.8150	172.9559
	17:28:23	0:52:05	9.4755	23.8502	172.9506
11-Nov-96	16:04:47	0:56:20	9.3549	24.5143	172.8633
	17:14:37	0:52:35	9.3473	24.5515	172.8567
12-Nov-96	15:46:14	0:57:03	9.2314	25.2983	172.6072
	16:58:15	0:53:12	9.2246	25.3382	172.5995
13-Nov-96	15:24:50	0:57:56	9.1257	26.1236	172.3150
	16:39:57	0:53:55	9.1198	26.1667	172.3063
14-Nov-96	15:02:16	0:58:55	9.0466	26.8873	172.1402
	16:15:21	0:54:55	9.0287	27.1824	171.7576
15-Nov-96	14:29:58	1:00:22	8.9811	27.7543	171.8299
	15:47:27	0:56:08	8.9725	28.1834	171.3326
16-Nov-96	18:36:33	0:47:18	8.9162	28.7341	171.5825
17-Nov-96	18:21:17	0:47:52	8.8939	29.6635	171.2614
18-Nov-96	18:07:04	0:48:25	8.8888	30.4871	171.0732
19-Nov-96	17:52:40	0:48:58	8.9003	31.2739	170.9072
20-Nov-96	17:34:27	0:49:42	8.9336	32.2347	170.6113
21-Nov-96	17:17:37	0:50:23	8.9721	33.0381	170.4474
22-Nov-96	16:59:17	0:51:08	9.0213	33.8373	170.2947
23-Nov-96	16:33:04	0:52:16	9.1017	34.8534	170.0216
24-Nov-96	16:05:44	0:53:27	9.1758	35.6671	169.8820
25-Nov-96	15:09:45	0:56:02	9.2576	36.4706	169.7374

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4. Cruise

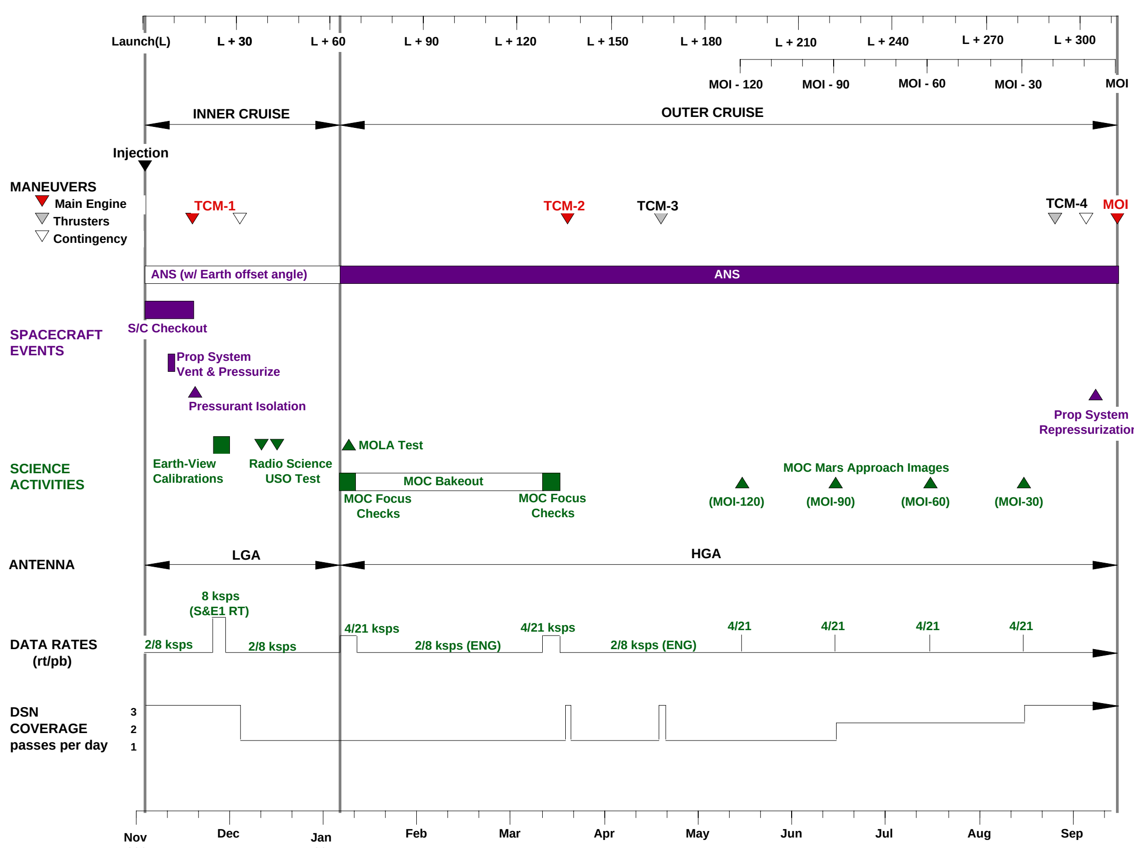
Trans-Mars cruise covers the time of ballistic flight between Earth and Mars. The spacecraft will take between 301 and 309 days to reach the red planet on its Type-2 trajectory depending on the Earth departure date within the 20-day launch period. A launch at the open of the launch period on 6 November 1996 will correspond to a Mars arrival date of 11 September 1997, while a launch at the close of the period on 25 November 1996 will result in an arrival on 22 September 1997.

During cruise, a set of four trajectory correction maneuvers (TCMs) will adjust the interplanetary trajectory to ensure that the spacecraft reaches the proper velocity and position targets prior to the Mars orbit insertion (MOI) burn. Other primary activities during cruise will include daily monitoring of the subsystems and science instrument checkout and calibration activities. Figure 4-1 shows an overall timeline for the cruse phase based on a 6 November 1996 launch date.

4.1 Initial Deployment and Acquisition

The cruise phase of the mission will start immediately after spacecraft separation from the third stage of the Delta 2. Separation from the booster will trigger a command sequence that will initiate deployment of the solar arrays and turn on the spacecraft's transmitter to allow for acquisition from the tracking antennas of the Deep Space Network (DSN). This sequence represents the most critical mission period for two reasons. First, spacecraft survival will critically depend on proper deployment of the solar arrays because the spacecraft can only operate on battery power for a limited amount of time after launch. Sec-

Figure 4-1: Cruise Timeline



ond, initial acquisition of the spacecraft's signal by the DSN must occur within about 100 minutes after separation because the X-band acquisition-aid antenna can only receive the spacecraft's signal to a range of about 40,000 km.

The acquisition-aid antenna is a small, wide-beam dish mounted on the 26-meter antenna at the Canberra tracking site. This small antenna will initially direct pointing of the larger, narrow-beam, 34-meter high-efficiency tracking antenna (34m HEF) that will normally be used to track the spacecraft. Not acquiring the spacecraft's signal will prevent the navigation team from gathering the initial two-way Doppler data necessary to determine the spacecraft's trajectory for the purpose of producing an accurate ephemeris. In turn, lack of an accurate ephemeris will dramatically hamper the ability of the 34m HEF to find the spacecraft.

4.1.1 Initial Deployment

Before the spacecraft separates from the Delta third stage, a yo-yo cable device attached to the stage will deploy to reduce the spin rate from 60 r.p.m. down to zero with a plus or minus two r.p.m. uncertainty. The device works by transferring the angular momentum of the third stage and spacecraft to the cable. During the despin process, redundancy management will be disabled to prevent the computer logic from incorrectly identifying possibly-saturated gyros as failed in the unlikely event that rotational rates exceed 9^{o} per second (1.5 r.p.m.) after despin.

Five seconds after yo-yo release, pyrotechnic devices will fire to sever the connection between the spacecraft and third stage. A set of four springs will then uncoil to impart a relative separation velocity of between 0.6 and 2.4 m/s between the third stage and spacecraft. The actual mission elapsed time of separation will vary as a function of launch day between the values of 43 and 56 minutes, but will always occur 370 seconds after third stage ignition. For a launch at the open of the period on 6 November 1996, separation will occur about 50 minutes after launch. Table 4-1 shows a listing of the post-separation events.

A set of breakwires attached to the Delta's third stage will provide a signal to the spacecraft that separation has occurred. Once this signal triggers, the post-separation command sequence will activate. First, the attitude-control thrusters will be armed and enabled in the first two seconds after separation detection, allowing the launch despin control mode to be activated as soon as possible. The despin control software will "snapshot" the current attitude and fire the appropriate thrusters to despin the spacecraft and hold attitude. This process will take no more than two minutes under normal circumstances. After this two minute time period, the thrusters will be disarmed in preparation for solar array deployment.

Following thruster disarming, the two folded solar arrays will deploy one after the other by a release of each array's outer and inner panel in that order. Once released, the solar arrays will unfold to their fully deployed configuration by four pairs of spring-driven hinges. After each hinge rotates approximately 180^{o}, a latch will engage and lock the hinge and panel in place. In the timeline, five minutes has been allocated for the solar array deployment process. In addition, another 20 seconds has been allocated for the attitude control thrusters to remove any incidental spacecraft rotation introduced by the release of the solar panels.

At this time the spacecraft will begin to acquire the attitude required for DSN initial acquisition. Because the spacecraft will be spinning when it separates from the Delta, it will not have a three-axis attitude reference. The only attitude knowledge the spacecraft will possess is the location of the Sun as determined from the Sun sensors (SSA). For DSN initial acquisition, the spacecraft will enter the Sun acquisition and coning mode, called SUN-COM-POWER, in which the +X axis will be aligned to the measured Sun vector. In the timeline, five minutes has been allocated for the spacecraft to slew until the SSA detects the Sun, and an additional three minutes has been allocated to point the +X axis at the Sun. Under normal circumstances, the spacecraft will enter the DSN initial acquisition attitude within 18 minutes of spacecraft

Table 4-1: *Post-Separation Timeline*

Event Description	Time From Separation	Comments
Third stage burn out	-370 seconds	Actual time varies by launch date (between L+ 39 to 52 minutes)
Yo-yo deploy	-5.0	Despin spacecraft and third stage from 60 r.p.m.
Third stage jettison	0.0	Breakwires attached to third stage provide separation signal to spacecraft
Arm thruster string 1 & 2	0.1	Thruster enable commands already issued during parking orbit coast
Begin thruster firings	0.6	Remove any residual rotation not removed by yo-yo device
Disarm thrusters	126.6	Prevents external torques during solar panel deployment
Initiate solar panel deploy	127.6	Five minutes allocated for deploy
Exit eclipse (penumbra)	159.3	Value for 6-Nov, maximum value of 835 seconds for 25-Nov launch
Outer solar array gimbal begin	428.6	Gimbal from -85^o to +30^o at 0.7^o/sec
Inner solar array gimbal begin	471.5	Gimbal from +90^o to -15^o at 0.7^o/sec, begin when outer gimbal at -55^o
Turn on star sensor (CSA)	479.7	CSA turned on 30 seconds prior to use
Arm thrusters	478.6	Delay from array deploy begin time to keep panels out of thruster plume
Begin null attitude rates	488.6	Two minutes allocated to remove spin due to solar array deployment
Initiate payload Sun avoidance	509.6	Enable autonomous Sun avoidance logic for payload protection
Begin Sun acquisition	509.7	Eight minutes allocated to find Sun and point +X axis at Sun
End Sun acquisition	989.7	Roll at 0.06^o/sec, will occur later if eclipse exit has not yet occurred
Turn on TWTA filament	1050.9	Allow four minutes for warm-up
Turn on TWTA beam	1290.9	Beam power will use 55 Watts of power
Turn on transponder (MOT)	1291.9	Begin DSN initial acquisition period of two hours
Move to Sun coning orientation	8491.9	Begin slew to +X axis 60^o off vector to Sun
Begin Sun coning	8691.9	Begin coning at 0.06^o/sec (100 min/rev)
Begin SUN-STAR-INIT	8899.0	Use CSA to establish attitude reference (takes about 200 minutes)

separation. At that time, the spacecraft will begin to spin about its +X axis at a rate of one revolution every 100 minutes.

During the transition from the post-despin attitude to the DSN initial-acquisition attitude, the solar panels will automatically be commanded to their cruise orientation by rotating the outboard (azimuth) gimbals by 120^o and the inboard (elevation) gimbals by 90^o. In this configuration, the solar arrays will be swept forward toward the Sun, 30^o above the Y axis in the +X direction. Approximately 12 minutes after separation, four minutes prior to the spacecraft achieving the DSN initial-acquisition attitude, the filament of the TWTA configured to the primary transmit low-gain antenna (LGA) will be activated for warm-up.

4.1.2 DSN Initial Acquisition

Initial acquisition will begin about 18 minutes after the spacecraft separates from the Delta's third stage. At this time, the post-separation command sequence will command the spacecraft to begin transmitting realtime engineering data over the LGA at a rate of 2,000 bps. This transmission rate will allow the ground-control team to instantly determine whether the spacecraft entered safe mode prior to initial acquisition. The reason is that if safe mode entry occurs during launch, then the first transmission will appear at the slower, safe mode utilized rate of 10 bps.

Use of the short coast launch trajectory will place the spacecraft over the Canberra tracking site for initial acquisition. The DSN estimates that using the listen only, wide-beam, X-band acquisition-aid (ACQ-AID) antenna, they will "lock-up" on the carrier portion of the signal within a few minutes after the spacecraft begins transmitting. After detection of the carrier, the DSN will listen to the downlink signal for about TBD minutes to "fine tune" the best uplink frequency with which to establish a coherent, two-way lock.

In order to establish two-way, the DSN will track the spacecraft downlink with the ACQ-AID and use its pointing data to point the narrow-beam, 34m HEF antenna. Once the 34m HEF locks onto the spacecraft's signal, the ACQ-AID will no longer be needed. The DSN estimates that under normal circumstances, establishing a coherent, two-way lock with the spacecraft will require up to 30 minutes after the detecting the signal on the ground.

Under all circumstances, signal lock-up with the 34m HEF must occur within 105 minutes of the time that the spacecraft begins to transmit telemetry to the ground. After 105 minutes elapse, the spacecraft's range to the Canberra tracking site will exceed 40,000 km, a distance greater than the ACQ-AID's specified "listen range" given the spacecraft's transmission link margin. If 105 minutes elapse and lock-up has not yet occurred, the 34m HEF can perform initial acquisition. However, such a task will be extremely difficult because of the HEF's narrow beamwidth.

During the initial acquisition period, the spacecraft will maintain its orientation of +X axis pointed directly at the Sun, solar panels swept forward 30^o above the Y axis in the direction of +X, and roll rate of one revolution every 100 minutes about the +X axis. The spacecraft will continue to hold this attitude for a total of two hours starting from the time that LGA begins transmitting. This time period exists primarily to allow the navigation team to collect at 1.5 hours of two-way, coherent Doppler data for orbit prediction purposes. In addition, the ground operations team will examine the realtime engineering telemetry to assess the health and status of the spacecraft.

The DSN multi-mission navigation team will require at least 1.5 hours of coherent, two-way Doppler data in order to guarantee an orbit prediction accurate enough to point the 34m HEF antenna at the spacecraft to within a half beamwidth. In the current DSN initial acquisition timeline, the first 30 minutes that elapse after the transmitter turns on has been allocated to establishing a two-way lock on the spacecraft. Therefore, the view geometry from the LGA to the ground must support a viable downlink until two hours after the transmitter turns on (2 hours, 22 minutes after separation).

4.1.3 Attitude Initialization

Two hours after the start of the DSN initial-acquisition period (142 minutes after separation), the spacecraft will begin its attitude-initialization sequence. In order to establish a 3-axis attitude reference, the spacecraft must scan its celestial star sensor (CSA) around the sky to identify known stars. This attitude-initialization process will be accomplished by pointing the spacecraft +X axis somewhere along the edge of an imaginary cone with a half-angle of 60^o and longitudinal axis along the vector to the Sun. In this orientation, the spacecraft will roll so that its +X axis traces a path around this 120^o Sun exclusion cone once every 100 minutes. Therefore, the spacecraft +X axis is said to be "coning 60^o off the Sun." The coning motion will last for several 100-minute revolutions to allow the star sensor (CSA) enough time to acquire a three-axis attitude reference.

While the spacecraft cones around the Sun to perform attitude initialization, communications with the Earth will periodically fade in and out at 100 minute intervals. The reason is that during this time, the angle between the Sun and Earth, as seen from the spacecraft, will measure between 65^o to 105^o (depending on the launch day), and the extremes of the spacecraft's +X axis during the coning will place the axis plus or minus 60^o from the Sun. Because the LGA sits on the rim of the high gain antenna (HGA) and will point in the +X direction while the HGA sits in its stowed position, the LGA will cycle through positions of pointing almost directly at the Earth to pointing 125^o or more away from the Earth. Current analysis shows that if downlink is not guaranteed when the LGA points more than 90^o from the Earth, then downlink will only be available during 45 minutes out of each 100-minute coning revolution.

After attitude initialization has been completed, the ground team will command the spacecraft to inner-cruise orientation. In this mode, called array normal spin (ANS), the solar arrays will lie in the same position relative to the spacecraft body (30^o swept forward toward +X axis) as during Sun coning. However, the difference is that the +X axis will point 60^o off the Sun vector in the direction of the Earth, and the spacecraft will roll at a rate of one revolution every 100 minutes around the +X axis.

Spinning 60^o off Sun represents a compromise between needing to point the +X axis directly at the Earth for maximum communications link margin, and needing to point the solar arrays at the Sun for adequate power generation. Communications with the Earth will always occur through the low gain antenna during inner cruise because the undeployed HGA on the +X axis must point directly at the Earth for use.

4.2 Trajectory Correction Maneuvers

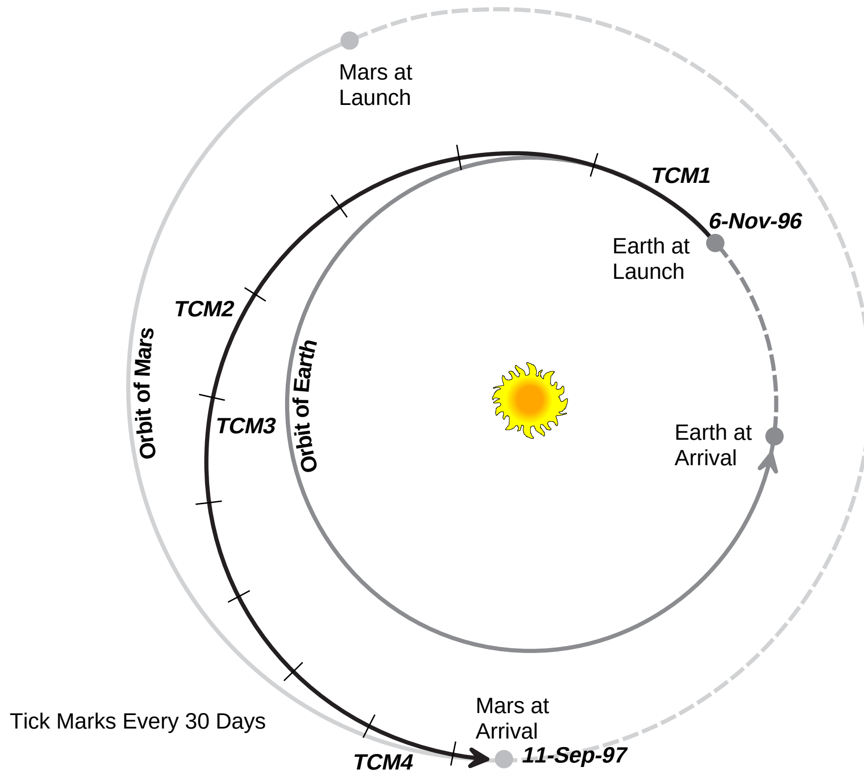
During cruise, a set of four trajectory correction maneuvers (TCMs) will adjust the interplanetary trajectory to ensure that the spacecraft reaches the proper velocity and position targets prior to the Mars orbit insertion burn. In general, the TCMs will predominantly be statistical maneuvers to correct for injection errors from the Delta third stage, orbit determination errors, unmodelled forces, and slight execution errors from previous TCMs. In addition, the TCMs will remove the Mars aim-point biasing introduced by the Delta for planetary-protection purposes. This launch biasing intentionally aims the spacecraft away from Mars by about 50,000 km to guarantee a sufficiently-low probability of the Delta third stage impacting the planet. Figure 4-2 shows the locations of the TCM maneuvers during cruise.

Table 4-2 shows the TCM schedule along with the expected magnitudes for each of the four burns. Current project policy calls for budgeting the TCMs at a 95% confidence level instead of a 99% level due to tight ΔV budget situation. In the table, each row corresponds to a different TCM and contains two sets of values for the maneuver dates and burn magnitudes. The top set of numbers in each row reflects values for a launch on 6 November 1996, while the bottom set corresponds to values for a 25 November 1996 launch.

Table 4-2: ***TCM Schedule and Burn Magnitudes***

Maneuver / (type)	Time	Mean / (95%) Value	Comments
TCM1 (main engine - biprop)	L+ 15 days (21-Nov-96) L+ 15 days (10-Dec-96)	15.0 m/s (95% magnitude of 33.0 m/s) 21.1 m/s (95% magnitude of 41.7 m/s)	Correct for most of the injection errors, remove most of the launch biasing due to planetary quarantine purposes
TCM2 (main engine - blow down)	TCM1+ 120 days (21-Mar-97) TCM1+ 105 days (25-Mar-97)	4.1 m/s (95% magnitude of 5.7 m/s) 5.6 m/s (95% magnitude of 6.0 m/s)	Correct for execution errors from TCM1, remove remaining launch injection errors and planetary quarantine biasing
TCM3 (AACS - monoprop)	TCM2+ 30 days (20-Apr-97) TCM2+ 30 days (24-Apr-97)	0.087 m/s (95% magnitude of 0.151 m/s) 0.197 m/s (95% magnitude of 0.359 m/s)	Correct for errors from TCM2
TCM4 (AACS - monoprop)	MOI- 20 days (22-Aug-97) MOI- 20 days (2-Sep-97)	0.26 m/s (95% magnitude of 0.53 m/s) 0.26 m/s (95% magnitude of 0.50 m/s)	Final adjustment to MOI aim point
Total for TCMs		Launch on 6-Nov-96 (95% magnitude of 36.98 m/s) Launch on 25-Nov-96 (95% magnitude of 47.90 m/s)	Values for the TCM total represent the statistically combined total for the entire cruise phase at a 95% magnitude, not an algebraic sum. Also includes allowance for 95% launch vehicle PCS and revised spin and nutation time constants (59 r.p.m., 86 s)

Figure 4-2: ***Trans-Mars Cruise Trajectory Diagram and Location of TCMs***



The first and largest of the trajectory correction maneuvers, called TCM1, will always occur 15 days after launch. This maneuver will primarily correct for injection errors introduced by the Delta third stage and remove most of the launch biasing introduced into the trajectory for Mars planetary quarantine purposes. TCM2, scheduled for between 105 and 120 days after TCM1, will correct for errors in TCM1 and remove the remaining planetary-quarantine bias from the trajectory. Both of the first two TCMs will be performed with the main engine. However, the second TCM will be performed in “blow-down” mode with the residual pressure level remaining in the tanks after the isolation of the high-pressure helium line sometime after TCM1.

The last two correction maneuvers, called TCM3 and TCM4, will each consist of small burns designed to correct for slight execution errors in the previous maneuvers. In addition, TCM4 at 20 days prior to Mars orbit insertion (MOI) will precisely target the spacecraft to its MOI aim point. Both of these burns will utilize the 4.4-N attitude control thrusters.

Contingency maneuver windows will exist at L+ 30 days for TCM1 and at MOI- 10 days for TCM4 if the primary opportunities are missed due to unforeseen circumstances.

4.2.1 TCM Implementation

Both TCM1 and TCM2 will utilize the main engine (596-N thrust) for primary thrust output with the smaller engines (4.4-N thrust) for attitude control. The small magnitude of the other two maneuvers, TCM3 and TCM4, will allow the attitude control thrusters to perform the burns.

Before both TCM1 and TCM2, the spacecraft will turn from its normal cruise attitude and point the -Z axis (engine exhaust thrust axis) opposite the direction of the desired ΔV. In addition, the X and Y axes of the spacecraft in the maneuver attitude will point in a direction to optimize the Sun angle to the solar arrays. If this angle is too large, the spacecraft will need to operate on batteries during the maneuver.

Because TCMs are statistical maneuvers designed to correct errors in the trajectory, the spacecraft operations team will not know the exact direction and magnitude of the burn until several days before the scheduled maneuver date. Therefore, the turn may take up to 10 minutes if the magnitude of the turn angle measures as large as 180^o. After reaching the maneuver attitude, the spacecraft will then hold a fixed inertial attitude while autonomous AACS checks occur. The spacecraft will then execute the burn autonomously with the capability to abort in the event of a malfunction.

During all TCMs, the spacecraft will control the burn duration by using accelerometer measurements as inputs. At the end of the burn, the spacecraft will turn back to its original cruise attitude. For all maneuvers, at least one of the recorders will continuously record engineering data for subsequent playback. In the case of TCM1, the spacecraft will be close enough to the Earth to also return data at the 2 ksps engineering rate in realtime, given a favorable maneuver direction in terms of sunlight on the solar array and low gain antenna (LGA) orientation with respect to the Earth. During the other three maneuvers, the link margin on the LGA will not support the 2-ksps engineering rate. Consequently, the transmitter will be shut off to conserve battery power. However, in the event that in-flight performance data shows an adequate power situation, the transmitter may be left on in “carrier only” mode during TCMs 2, 3, and 4.

Knowledge of the location of the Sun will factor heavily into the design of the TCMs because of a significant payload constraint on maneuver attitudes, required by the MOC and the MOLA instruments. These instruments do not have covers. Consequently, a flight rule exists that states that spacecraft's +Z axis (science instrument panel) cannot be pointed closer than 30^o from the Sun. Due to the fact that the ΔV direction for a TCM can be in any inertial direction, a performance penalty may exist for pointing the thrust in a non-optimal direction to satisfy the Sun pointing constraint.

4.2.2 Propulsion System Operations for TCMs

At launch the pressure in the entire propulsion system will sit at a blanket value of 100 psig. The mono-propellant lines will be launched wet down to the engine valves, allowing the attitude control thrusters to operate in blow-down mode to support spacecraft separation from the Delta and de-spin.

The main engine valve and latch valves below the tanks will remain closed at launch. Prior to TCM1, a dry-firing of the main engine for several minutes will vent the lines (between the latch and main engine valves) to space and bleed them dry. After venting, the ground team will command these latch valves to open. This action will allow the fuel and oxidizer to fill the lines up to the main engine valve at approximately the same blanket pressure used to operate the mono-propellant system in blow-down mode after launch.

Pressurization of the propulsion system will occur prior to TCM1. In order to pressurize the system, the high pressure latch valve located between the helium pressurant tank and the regulators will open, followed by the opening of one of the normally closed pyro valves located immediately below the pressurant tank. Tank pressurization will begin when the pressure downstream of the regulators reaches 175 psig and ruptures the two burst discs located upstream of the fuel and oxidizer tanks. Then, the regulator will come on line when the pressure in the tanks reaches the nominal value of 250 psig.

Due to concerns about regulator leakage with the tanks nearly full, one of the normally open pyro valves in the pyro ladder immediately downstream from the helium tank will be closed after TCM1, effectively isolating the high pressure helium from the rest of the system. Consequently, the remaining TCMs must occur in “blow-down” mode using the residual pressure level remaining in the tanks.

4.3 Inner Cruise Activities

Inner cruise will be devoted to characterizing the operation of the spacecraft, performing the first TCM, checking out the spacecraft, and calibrating the science instruments. Adequate link margins and continuous DSN coverage during this time period will support the return of data rates as high as the 40,000-sps, S&E-2 realtime rate for payload operations and check-out.

4.3.1 Major Spacecraft Activities and TCMs

Activities during the first half-month of flight will focus only on spacecraft check-out and preparations for executing the first trajectory correction maneuver (TCM) at 15 days after launch. No “special” check-out procedures exist other than planned intensive monitoring of each subsystem with contingency recording of telemetry on one of the recorders. In addition, telemetry recorded during launch will be replayed during this time.

Propulsion system activities will begin at L+ 7 days with the “dry-firing” of the main engine. The “dry-firing” will vent the lines (downstream of the tanks) to space and bleed them dry. Pressurization of the tanks with the high pressure helium will follow on flight day L+ 12 in preparation for the first TCM at L+ 15 days. This schedule implementation will provide margin before the pressurization and TCM in the event of an anomaly during the first few days of flight.

Upon successful completion and verification of TCM1, one of the pyro valves immediately downstream of the helium tank will be closed to effectively isolate both the oxidizer and fuel tanks from the high pressure assembly upstream. Re-pressurization of the tanks will not occur until just prior to Mars orbit insertion (MOI).

4.3.2 Instrument Calibrations

Instrument calibrations will begin on flight day L+ 16 with the activation and checkout of the payload data subsystem (PDS). This device is the computer control unit that collects data from the instruments and formats the information into data packets for storage on the SSRs or playback to Earth. The checkout sequence will include a flight software upload into the PDS RAM. Following the software upload, the different data-rate modes controlled by the PDS will be tested by broadcasting data to Earth at all of the S&E1 and S&E2 data rates from 4 ksps to 85 ksps. Previous experience has shown that the software upload and data-rate test will consume most of the prime-shift working day.

On day L+ 17, the flight team will activate all of the science instruments and will provide a brief time period for the science investigation teams to send flight-software uploads to their instruments. Later in the day, the spacecraft will perform an hour-long slew to allow the TES to image the Earth and the Moon. These images will allow the TES team to calibrate their instrument.

Later on in the day, the radio science team will perform a checkout of the ultra-stable oscillator (USO). This test will involve turning off the spacecraft’s telemetry and broadcasting to Earth in carrier-only mode for two hours. The USO test will allow the radio science team to determine the frequency stability characteristics of the device. Two more USO tests will follow in mid-December 1996.

In addition, the Mars Relay (MR) will begin broadcasting its UHF relay beacon on day L+ 17. The MR test will consist of two parts. The primary portion of the test will involve use of the 46m antenna at Stanford to broadcast test data to the MR for storage. The spacecraft will store this data on its SSRs for subsequent playback to Earth. The secondary portion of the test will involve amateur radio enthusiasts around the world. These HAM operators will use their own radios to listen to MR’s signal beacon and report their results back to JPL.

The MOC will begin its calibration activities on flight day L+ 18. During the morning, the spacecraft will perform a one-hour slew to allow the camera to image the Earth and Moon for calibration purposes. Then, on flight day L+ 19, the spacecraft will perform another one-hour slew for the MOC to image several stars. These two days worth of images will allow the MOC team to make minor focus adjustments by altering heater settings on the camera.

4.3.3 *Important Modification to Baseline Schedule*

For the first seven days of the launch period (November 6th through November 12th, inclusive), TCM1 and isolation will take place on flight days L+ 14 and L+ 15, respectively. Executing the maneuver *one day earlier* than in the baseline plan will allow the instrument calibration activities (L+ 16 to L+ 19)to finish by the Thanksgiving holiday.

4.4 Outer Cruise Activities

About two months after launch and sometime in January 1997, the angle between the Earth and Sun as seen from the spacecraft (values of 60^o or less) will allow for pointing the undeployed high-gain antenna (HGA) directly at Earth and still provide for adequate illumination of the solar arrays. This geometry will allow the spacecraft to transition from using the LGA to the HGA.

From this time until Mars orbit insertion, excluding trajectory correction maneuvers, the normal spacecraft configuration will be outer-cruise array normal spin (ANS) in order to take advantage of the decreasing angles between the Earth and Sun. When configured in this mode, the spacecraft solar panels will lie swept forward 30^o above the Y axis toward the +X direction, the +X-axis of the spacecraft will point directly at the Earth, and the spacecraft will roll about the +X axis at the rate of one revolution every 100 minutes.

4.4.1 Normal Spacecraft Cruise Operations

Primary activities during outer cruise will involve routine monitoring of the spacecraft, collection of navigation data, and execution of the remaining three trajectory correction maneuvers. Because the spacecraft HGA will point directly at Earth during outer cruise, substantial capability will exist for returning data from the science payload. However, only a limited number of science data collection activities and calibrations are currently planned due to the limited staffing level of the flight team.

Normal operations of the spacecraft during cruise will consist of fairly uncomplicated tasks. Except for TCMs and limited science activities, minimal commanding will be sufficient to maintain the engineering sub-systems. The primary ground activity will involve analyzing trends in the performance of various sub-systems based on engineering telemetry and radio-metric data returned during the daily DSN tracking pass.

Although the spacecraft can operate for many days without contact, a command-loss timer will initiate fault protection responses if a command fails to arrive within a set amount of time. This command-loss time period is variable and can be changed. The only routine uplinks required to maintain the spacecraft will consist of “no-op” command-loss timer reset commands, and a bi-weekly star catalog and planetary ephemeris update. Data from this ephemeris will assist in the pointing of the spacecraft’s +X axis in the array-normal-spin attitude mode, while the star catalog will be used to update and maintain the spacecraft’s inertial reference.

4.4.2 Major Spacecraft Activities and TCMs

Until preparations for Mars orbit insertion (MOI) begin toward the end of outer cruise, no major spacecraft activities will occur other than the execution of TCM2, TCM3, and TCM4. Due to the isolation of the high-pressure helium tanks after TCM1, the three remaining maneuvers will occur in “blow-down” mode using the residual pressure level remaining in the fuel and oxidizer tanks. Seven days prior to MOI, the propulsion system will be repressurized to 250 psig by firing one of the normally-closed pyro valves immediately upstream of the regulator system. Seven days will allow margin in the event of an anomaly associated with the repressurization of the propulsion system.

4.4.3 Major Payload Activities

Instrument checkout activities will continue in early-January 1997 with the MOLA test. During this one-hour activity, the spacecraft will slew its +Z axis toward the Earth to allow MOLA to fire laser pulses at the Earth. The MOLA team will utilize ground stations equipped with laser-detection devices to verify proper operation of the laser altimeter.

One day after the laser test, the MOC will resume its focus testing. These tests will occur on four separate occasions spaced one or two days apart. Each test will consist of a single one-hour spacecraft slew to allow the camera to image several stars. The images of the stars will allow the MOC team to make minor focus adjustments by altering heater settings on the camera.

After performing the four focus-check tests, the MOC will enter a 60-day bakeout period. During this time, a special set of heaters in the camera will continuously “bake” the instrument at low heat levels to solidify the desired focus adjustments. Upon completion of the bakeout event, the camera will perform another focus-check test. These post-bakeout focus checks will be performed in a manner similar to the pre-bakeout focus test.

In addition to performing focus checks, the MOC will acquire approach images of Mars at various distances prior to arrival. The first of four imaging opportunities will take place 120 days prior to Mars orbit insertion (MOI). The remaining three will occur at MOI- 90, 60, and 30 days.

The radio science team will also conduct tests of the USO in outer cruise. The tests will be similar to the one conducted on flight day L+ 17 and will occur approximately once every two to four weeks. Exact dates will be announced during cruise.

5. Orbit Insertion Phase

The orbit insertion phase of the mission will begin with the Mars orbit insertion (MOI) burn in mid-September 1997. This burn will slow the spacecraft by approximately 980 m/s and allow Mars to capture it into a highly elliptical orbit with a 48-hour period. Unfortunately, the MGS spacecraft will not carry enough propellant to propulsively reach the required low-altitude, Sun-synchronous mapping orbit due to the relatively low interplanetary injected mass capability of the low-cost Delta booster.

Instead, the spacecraft will rely on aerobraking, an innovative mission-enabling technique, to trim the initial, highly elliptical, capture orbit down to mapping orbit altitudes. During aerobraking, the spacecraft will pass through the upper fringes of the Martian atmosphere on every periapsis pass. Air resistance from the atmosphere during the "drag pass" will cause the spacecraft to lose a small amount of momentum and will cause the altitude on the next apoapsis pass to slightly decrease.

Aerobraking will end in mid-January 1998. Over the course of the next two months, the spacecraft will use a combination of propulsive maneuvers and orbital perturbations from the Martian gravity field to reach the desired mapping orbit. The orbit insertion phase will end with the beginning of mapping in mid-March 1998. Figure 5-1 shows a timeline of the events that occur during orbit insertion.

5.1 Mars Orbit Insertion

MOI will slow the spacecraft and allow Mars to capture it into an elliptical orbit. Before the burn, the spacecraft's velocity relative to Mars will measure approximately 5,650 m/s. Near the closest approach

Figure 5-1: ***Orbit Insertion Timeline (open of launch period scenario)***

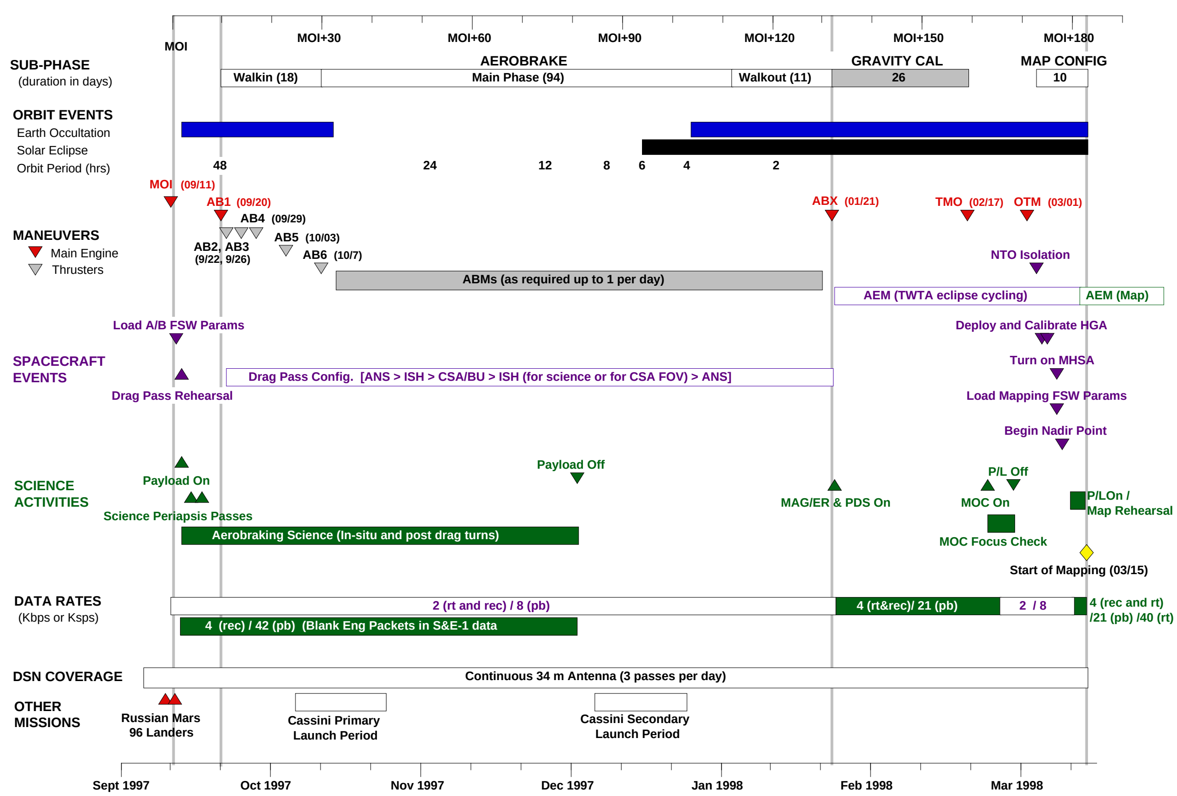
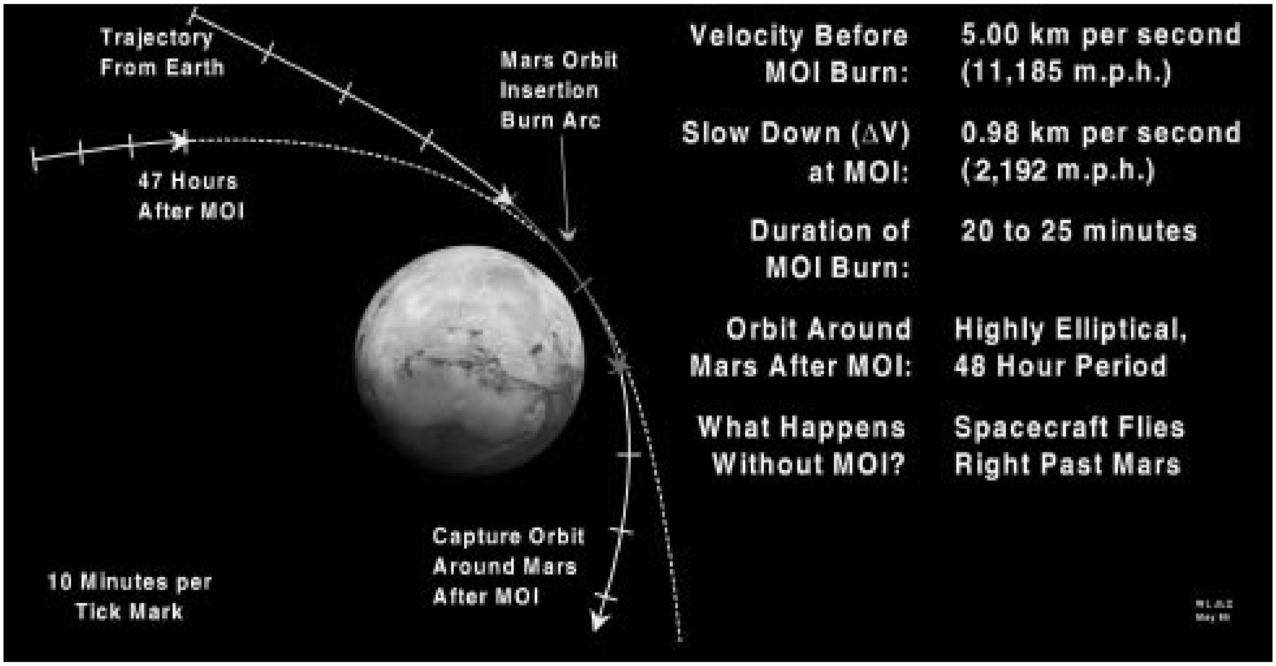


Figure 5-2: ***Trajectory Plot of MOI***



point of the inbound hyperbolic trajectory, the 659-N main engine will fire for between 20 to 25 minutes to provide a ΔV of about 980 m / s. Burn ignition will occur approximately about 10 minutes before Mars closest approach. Figure 5-2 shows a trajectory plot of the maneuver.

During the burn, the spacecraft will utilize a “pitch-over” maneuvering strategy to slew the spacecraft’s attitude at a constant rate in an attempt to keep the thrust nearly tangent to the trajectory arc. Although this constant rate pitch will not allow the thrust vector to exactly follow an optimal steering profile, the strategy will provide a more optimal solution than a constant burn vector pointed in a fixed, inertial direction.

After MOI-burn cut-off 10 minutes after periapsis, the spacecraft will orbit Mars on a highly elliptical orbit with a period of 48 hours and periapsis altitude of about 300 km (periapsis radius of 3,700 km). Due to specification uncertainties for maneuver execution, the capture orbit period may vary from as low as 40.3 hours to as high as 59.7 hours. In addition, the periapsis altitude may vary from 229 km to 399 km. Both the period and altitude figures represent 3σ dispersions due to specification uncertainties. However, the true capability errors will be smaller and will cause these uncertainty figures to be reduced.

5.1.1 Maneuver Execution Time

Depending on the exact launch date, the spacecraft will arrive at Mars between 11 September 1997 and 22 September 1997. The incoming hyperbolic trajectory will be targeted for a Mars closest approach at roughly 1:00 a.m. UTC to take advantage of overlapping DSN coverage from Goldstone and Canberra. The exact time of Mars closest approach will vary depending on the arrival date, but will always occur approximately half-way into the overlap period between the two DSN stations. This timing strategy will provide about one hour of overlapping tracking coverage both before and after the MOI burn.

Goldstone will serve as the primary station for pre-MOI burn coverage, while Canberra will take over for post-burn tracking. This decision was based on allowing the station complex with the higher elevation angle view of Mars to be prime for uplink. However, both the 34m HEF and 70m antennas at both complexes will listen to the spacecraft’s signal during the overlap period. If the 34m HEF at the prime complex fails, then the 70m will be able to automatically take over in listen-only mode. In the event that commanding capability or an uplink carrier is required, then prime tracking responsibility will switch to the 34m HEF at the other complex.

Table 5-1 lists the arrival date and time of Mars closest approach for each launch date. Under normal conditions, the MOI burn will be centered on the closest approach and will last for between 20 to 25 minutes. In the table, the closest approach and Goldstone set (DSN overlap end) occurs on the same calendar day as the arrival day. The Canberra rise time (DSN overlap begin) occurs on the calendar day before the arrival day. All times listed in the table are in ephemeris time (ET).

Table 5-1: ***MOI Times***

Launch Date	Arrival Date	Canberra Rise	Closest Approach	Goldstone Set
6-Nov-96	11-Sep-97	23:43:50	01:27:53	03:40:32
7-Nov-96	12-Sep-97	23:42:02	01:26:09	03:38:32
8-Nov-96	12-Sep-97	23:42:02	01:26:09	03:38:32
9-Nov-96	13-Sep-97	23:40:15	01:24:28	03:36:33
10-Nov-96	13-Sep-97	23:40:15	01:24:28	03:36:33
11-Nov-96	14-Sep-97	23:38:29	01:22:47	03:34:35
12-Nov-96	14-Sep-97	23:38:29	01:22:47	03:34:35
13-Nov-96	14-Sep-97	23:38:29	01:22:47	03:34:35
14-Nov-96	15-Sep-97	23:36:44	01:21:07	03:32:38
15-Nov-96	15-Sep-97	23:36:44	01:21:07	03:32:38
16-Nov-96	16-Sep-97	23:35:00	01:19:29	03:30:42
17-Nov-96	16-Sep-97	23:35:00	01:19:29	03:30:42
18-Nov-96	17-Sep-97	23:33:17	01:17:52	03:28:48
19-Nov-96	18-Sep-97	23:31:35	01:16:14	03:26:54
20-Nov-96	18-Sep-97	23:31:35	01:16:14	03:26:54
21-Nov-96	19-Sep-97	23:29:54	01:14:40	03:25:02
22-Nov-96	20-Sep-97	23:28:15	01:13:05	03:23:10
23-Nov-96	20-Sep-97	23:28:15	01:13:05	03:23:10
24-Nov-96	21-Sep-97	23:26:36	01:11:32	03:21:20
25-Nov-96	22-Sep-97	23:24:59	01:10:01	03:19:30

MOI Time Changes for Russian Lander Support

During the four orbits immediately following MOI, the spacecraft may be required to relay data back to Earth from Russian landers on the surface by using the Mars Relay antenna. Unfortunately, the MOI times chosen for the dual-station overlap strategy will result in a ground track pattern incompatible with overflights of Russian landing sites after MOI. If relay operations are required, then the MOI time will be changed at TCM2 to produce a post-burn ground track that overflies the landing sites within the first four orbits after the burn.

Initial analysis shows that MOI will need to be moved between two and seven hours earlier to facilitate the relay of Russian data. Calculations of the exact MOI offset time will be performed after both the MGS spacecraft and the Russian landers have launched during the November 1996 opportunity. In general, MGS launches early in the launch period will result in a greater offset as compared to launches later in the launch period.

Moving MOI earlier will result in two major impacts to the mission. First, dual-station overlap for pre-burn and post-burn tracking redundancy will be lost because the Madrid to Goldstone overlap will last for less than 30 minutes in September 1997. Second, changing the time of MOI at TCM2 will cost between 3 to 6 m/s. The ΔV penalty will have substantial ramifications to the ability to alter the MOI time because TCM2 will be limited to 6 m/s or less. This limitation results from the fact that TCM2 will be performed in “blow-down” mode using the residual pressure level remaining in the tanks following isolation of the high-pressure helium lines after TCM1.

Altering the MOI time at TCM1 will not be possible due to the fact that the Russian launch period opens on 16 November 1996. Due to a high desire to maintain the dual-station overlap during MOI strategy, the project will not commit to altering the Mars arrival time until it has been verified that the Russians have launched. Although November 16th will occur before TCM1, sufficient time will not exist to incorporate the navigational targeting changes into the flight sequence before the execution of the maneuver.

5.1.2 Spacecraft Activities During MOI

The final phase of the MOI burn sequence will begin with the start of 2-kbps engineering telemetry recording on two of the recorders. Prior to the start of the burn, the spacecraft will power on the catbed and main engine heaters, begin the turn to the pre-determined burn attitude using reaction wheel control, and then gimbal the solar arrays to their maneuver positions by rotating the outer gimbals -90^{o} from their cruise configuration. At this time, the TWTA beam will be turned off both to conserve power and because the spacecraft will be occulted by Mars as viewed by Earth throughout much of the burn.

At the pre-specified burn start time, the flight software maneuver task will fire the 659-N main engine. During the burn, the spacecraft will execute a “pitch-over” steering strategy to maximize the ΔV efficiency of the maneuver. This strategy will be implemented by using the attitude control thrusters to slew the spacecraft at a fixed rate during the burn in an attempt to keep the thrust tangent to the trajectory arc. In addition, the thrusters will provide attitude control during the burn.

The MOI maneuver will terminate after the spacecraft’s accelerometers have sensed the proper amount of velocity change. Termination will normally occur between 20 to 25 minutes after ignition. However, the spacecraft will automatically cut-off the engine if the burn duration exceeds the maximum allowable time. This maximum time value is a changeable parameter that will be pre-determined by the flight team prior to the loading of the MOI sequence.

Immediately after the completion of the burn, the spacecraft will begin the slew back to ANS Earth-point configuration. Then, downlink telemetry will be turned on and 2-kbps, realtime engineering data will be broadcast to Earth. The realtime telemetry will allow the flight team to assess the post-burn health status of spacecraft and will provide tracking data for a navigation solution to the capture orbit.

The spacecraft has several levels of fault protection that can impact the success of the MOI maneuver. For example, redundancy management software will be enabled for all critical hardware components. Upon detection of a fault, the spacecraft will automatically switch to the redundant hardware. The spacecraft will also use special system-level fault-protection modes, designated as safe mode and contingency mode. If entered, these modes will abort the MOI sequence. Consequently, safe and contingency modes will be disabled at appropriate times prior to the start of the maneuver. These times will be determined by the estimating the recovery times needed to reestablish the spacecraft to the required configuration for MOI.

5.2 Capture Orbit Activities

After successful completion of the MOI burn, the spacecraft will orbit Mars once every 48 hours with a periapsis and apoapsis altitude of approximately 300 km and 56,675 km, respectively. For the next nine days, the spacecraft will remain in this capture orbit as the flight team prepares for the beginning of aerobraking. Activities during this time period will include aerobraking rehearsal orbits and the collection of a limited amount of science data. Table 5-2 provides an orbit by orbit summary of the activities that will take place during the post-MOI orbits.

Table 5-2: ***Summary of Post-MOI Activities***

Spacecraft Position	Time	Activity	Configuration
First Orbit			
Periapsis #0	MOI	Mars Orbit Insertion	Maneuver
Apoapsis #1	MOI+ 1 day	Post-MOI Spacecraft Health Assessment	Array Normal Spin
Rest of Orbit	MOI to MOI+2	Post-MOI Spacecraft Health Assessment	Array Normal Spin
Second Orbit			
Periapsis #1	MOI+ 2 days	Routine Spacecraft Health Monitoring	Array Normal Spin
Apoapsis #2	MOI+ 3 days	Routine Spacecraft Health Monitoring	Array Normal Spin
Rest of Orbit	MOI+ 2 to MOI+ 4	Routine Monitoring, Prepare for Drag Rehearsal	Array Normal Spin
Third Orbit			
Periapsis #2	MOI+ 4 days	Aerobrake Drag Rehearsal	Drag
Apoapsis #3	MOI+ 5 days	Routine Spacecraft Health Monitoring	Array Normal Spin
Rest of Orbit	MOI+ 4 to MOI+ 6	Activate Science Instruments, PDS	Array Normal Spin
Fourth Orbit			
Periapsis #3	MOI+ 6 days	Science Data Collection	Nadir
Apoapsis #4	MOI+ 7 days	Routine Spacecraft Health Monitoring	Array Normal Spin
Rest of Orbit	MOI+ 6 to MOI+ 8	Routine Monitoring, Playback of Science Data	Array Normal Spin
Fifth Orbit			
Periapsis #4	MOI+ 8 days	Science Data Collection	Nadir
Apoapsis #5	MOI+ 9 days	AB1 Burn (lower periapsis to start aerobraking)	Maneuver
Rest of Orbit	MOI+ 8 to MOI+ 10	Routine Monitoring, Playback of Science Data	Array Normal Spin
Sixth Orbit			
Periapsis #5	MOI+ 10 days	First aerobrake drag pass (150 km altitude)	Drag
Apoapsis #6	MOI+ 11 days	AB2 Burn (lower periapsis some more)	Maneuver
Rest of Orbit	MOI+ 10 to MOI+ 11	Routine Monitoring	Array Normal Spin

5.2.1 Spacecraft Checkout

The navigation team will use radiometric data to track the spacecraft and determine the exact solution to the capture orbit during the first revolution around Mars. In addition, the flight team will monitor the engineering telemetry to determine the health and status of the spacecraft and its subsystems. This performance characterization will be necessary largely because of the spacecraft's entry into the new thermal environment around Mars.

During the second revolution around Mars, the flight team will prepare to conduct an aerobrake drag-pass rehearsal. The command sequences and scripts for aerobraking, as well as various flight software parameter updates, will be generated and uploaded prior to the second periapsis pass after MOI. Just before the passage through that periapsis, the spacecraft will configure itself into the proper attitude and orientation for flight through the upper Martian atmosphere. This rehearsal will test all of the ground operational procedures and spacecraft activities that will take place during aerobraking without subjecting the spacecraft to an actual drag pass.

5.2.2 Science Activities

Upon completion of the aerobrake drag-pass rehearsal at the start of the third orbit after MOI, the payload data subsystem (PDS) will be powered on and then checked out for the first time at Mars. Checkout activities will involve a flight software upload into the PDS RAM. Previous experience has shown that this upload will consume most of the prime-shift working day after the drag-pass rehearsal. Because the PDS controls the collection of data from the science payload, the instruments will not be powered on until after checkout of the PDS.

Six days after MOI, at the beginning of the fourth orbit, the spacecraft will slew to nadir point the science instruments at the surface of Mars for about 15 minutes centered at periapsis. During the 15 minutes, the MOC, TES, and MOLA will be able to record a limited amount of science. In addition, any use of the MR to collect data from Russian landers on the surface will also take place during this time. After the data collection period has expired, the spacecraft will return to array-normal-spin configuration and play back the recorded science at the highest available data rate. Science data collection will also be performed in this manner eight days after MOI at the beginning of the fifth orbit.

In order to minimize the effort required to execute the sequence of events required for periapsis data collection, the spacecraft will utilize a command script similar to the one that will be used during the aerobrake drag passes. Normally, the aerobraking command script temporarily slews the spacecraft from array normal spin (ANS) to drag pass configuration for a predetermined amount of time near periapsis. The only difference is that the spacecraft will fly with a 90^{o} offset in attitude during data collection. This offset will allow the science instruments on the +Z panel to point downward at Mars during periapsis instead of at the normal aerobraking attitude of +Z backward along the velocity vector.

From the time of instrument activation, the MAG, ER, and radio science (gravity field studies) will collect data on a continuous basis because they do not require nadir pointing. Upon completion of the two science-observation periapsis passes, the instruments will remain active for a significant portion of aerobraking in order to acquire unique science data and to support aerobraking operations.

5.3 Aerobraking

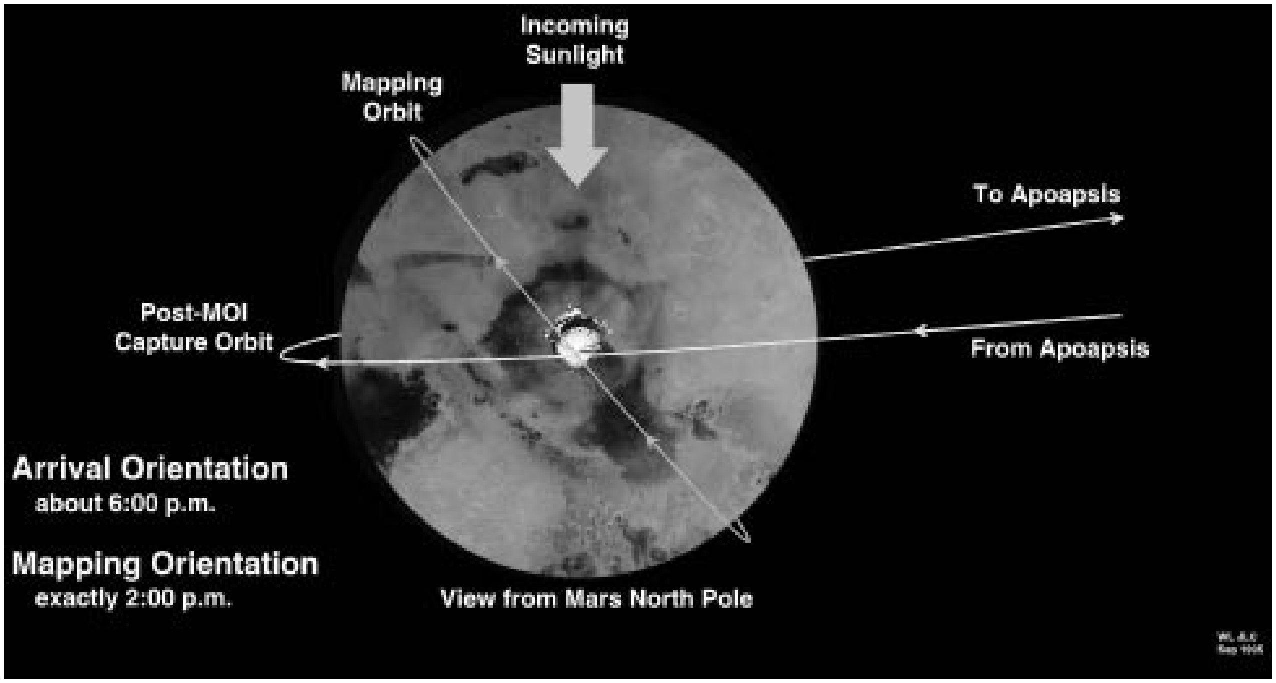
After completing the two “science orbits,” the spacecraft will spend the next four months aerobraking. This part of the mission will consist of three major phases called walk-in, main phase, and walk-out. During walk-in, the periapsis of the orbit will be gradually lowered into the Martian atmosphere at an altitude of about 112 km. Then, the spacecraft will enter main phase and remain there for the majority of aerobraking. In main phase, air resistance from the atmosphere will slow the spacecraft on every periapsis drag pass and will gradually cause the apoapsis to shrink. After the apoapsis altitude has decayed to less than 2,000 km, the spacecraft will use the walk-out phase to gradually raise the periapsis of its orbit out of the Martian atmosphere while continuing to finish aerobraking.

5.3.1 The General Scenario

One of the major constrains driving the aerobraking timeline is the requirement to reach a 2:00 p.m. descending node orientation with respect to the fictitious mean Sun. The spacecraft’s approach direction from its interplanetary trajectory will result in an initial node located at about 5:45 p.m. During most of aerobraking, the node time will move backward by about a half minute (0.524^{o}) per day due to Mars’ motion around the Sun.

Aerobraking must reduce the orbit size to mapping orbit altitudes in the exactly same amount of time that the node takes to move backward from 5:45 p.m. to 2:00 p.m. Because the nodal motion is to first-order independent of orbit size for most of aerobraking (56,675 km to about 10,000 km apoapsis), a “too-rapid” apoapsis decay profile will result in arriving at the mapping orbit with a node time later than 2:00 p.m. Conversely, a “too-slow” profile will result in a mapping orbit earlier than 2:00 p.m. Figure 5-3 illustrates the orbit node geometry constraints affecting aerobraking.

Figure 5-3: Orbit Insertion Geometry



5.3.2 Walk-in Phase

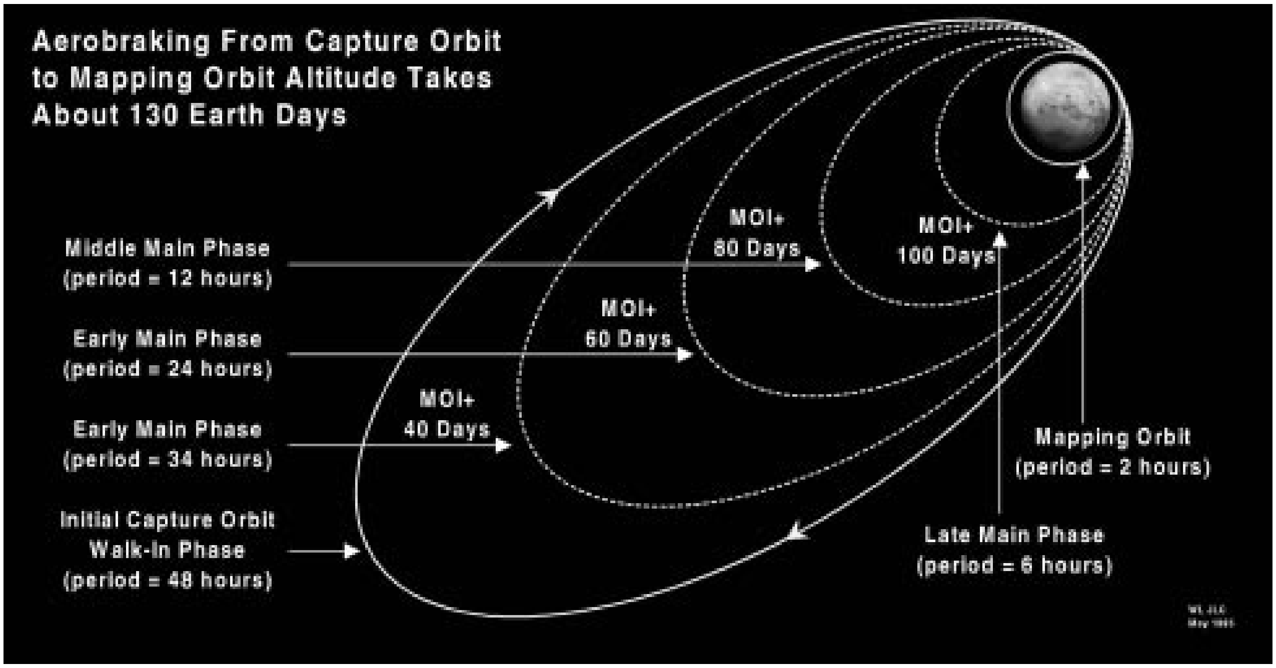
Aerobraking will begin nine days after MOI with a maneuver at the fifth apoapsis. This burn, called AB1, will be the first and largest of four to six mono-propellant maneuvers designed to lower periapsis into the upper Martian atmosphere in gradual steps. In addition, the burn will also correct for small inclination errors caused by MOI. After AB1, the spacecraft will take about one day to coast down to the new periapsis at an altitude of 150 km. The post-AB1 periapsis will represent both the start of the sixth orbit after MOI, and the first drag pass of aerobraking.

AB2 will occur at the sixth apoapsis and exactly one revolution after AB1. Afterward, the flight team will perform AB3, AB4, AB5, and AB6 at apoapsis on every other revolution. Therefore, AB6 will occur approximately 27 days after MOI at the 14th apoapsis. The goal is to use these six “walk-in” maneuvers to gradually lower periapsis to the target point where air resistance will cause the spacecraft to experience a heating rate of 0.38 W/cm^2. Current models predict that this heating rate will occur at an altitude of about 112 km and will amount to roughly 48% of the maximum tolerable rate of 0.79 W/cm^2.

Although AB1 will lower the periapsis to a pre-planned altitude, the navigation team will choose the size of the AB2 through AB6 burns in realtime during walk-in operations. The reason for the realtime selection is due to the large uncertainty in the atmospheric density model of Mars. In particular, the navigation team will need to estimate the scale height of the Martian atmosphere. This parameter quantifies the amount of altitude decrease needed to increase the atmospheric density by a factor of 2.718 (one exponential factor). Smaller scale heights indicate that the atmospheric density increases more rapidly as altitude decreases than for an atmosphere with a larger scale height number.

Unfortunately, post-drag-pass tracking data will yield the density of the atmosphere, but not the scale height. In order to determine the safety of a proposed walk-in maneuver, the navigation team will calculate a test parameter called the critical scale height. This parameter represents the scale height that would need to exist at Mars for the spacecraft to encounter the critical atmospheric density on the next drag pass after the proposed walk-in maneuver. The critical atmospheric density, defined as 143 kg/km^3, will cause the spacecraft to experience the maximum allowable drag-pass heating rate of 0.79 W/cm^2.

Figure 5-4: ***Phases of Aerobraking***



The navigation team will compare the critical scale height for the proposed walk-in maneuver to the scale height predicted by the Mars-Gram atmospheric model. If the critical scale height is less than the predicted actual scale height, then the proposed maneuver will probably not decrease the periapsis altitude into the “danger zone.” The reason is that a smaller critical scale height will indicate that the increase in atmospheric density with altitude decrease will need to be greater than predicted to cause the spacecraft to encounter the critical atmospheric density at the periapsis immediately after the maneuver.

During walk-in operations, the navigation team will select a burn magnitude from the list of 0.05, 0.10, 0.20, 0.40, 0.60, and 0.80 m/s for each one of the maneuvers from AB2 through AB6. The maneuver sizes for these five burns will be selected both to provide a gradual lowering of the periapsis altitude toward the target value at about 112 km, and to satisfy the critical-scale-height safety test. Because of the uncertainty in the knowledge of the Martian atmosphere, the target altitude that corresponds to the desired heating rate of 0.38 W/cm^2 will probably be different than predicted. Consequently, the navigation team may decide to change, delete, or add walk-in maneuvers as necessary.

5.3.3 Aerobrake Main Phase

After the completion of walk-in, the spacecraft will spend about 94 days in the main phase of aerobraking. During this time, repeated passes through the atmosphere at periapsis will cause the apoapsis altitude to shrink in size from about 56,675 km down to 2,000 km. As often as once per day, small propulsive maneuvers (ABMs) executed at apoapsis will maintain the periapsis altitude within a well-defined corridor low enough to produce enough drag to reduce the orbit within the time constraints to reach the 2:00 p.m. node, yet high enough to avoid spacecraft heating and maximum dynamic pressure limits. Due to the oblateness of Mars and the fact that periapsis will be migrating northward toward the pole during main phase, the altitude of periapsis will tend to rise. Consequently, most of the ABMs will be in the down direction to lower the periapsis altitude into the control corridor.

As compared to the walk-in and walk-out phases of aerobraking, spacecraft heating rates during the drag pass will reach a maximum during main phase. The spacecraft design can tolerate a top free-stream aerodynamic heating rate of 0.79 W/cm^2 at periapsis without violating the qualification limits of the solar arrays. Given the amount of time that the node takes to move from 5:45 p.m. to 2:00 p.m., the spacecraft will be able to aerobrake at an apoapsis decay rate that corresponds to a maximum drag-pass heating rate of 0.38 W/cm^2. Because atmospheric density is related to the heating rate, this baseline plan will provide margin for an orbit-to-orbit atmospheric variability of up to 90% (70% of true variability, 20% for navigation uncertainty) without violating the spacecraft's heating limits.

5.3.4 Walk-Out Phase

The two weeks of aerobraking following the main phase will represent an extremely critical period with respect to maintaining a viable orbit as the spacecraft lowers its apoapsis toward the target finish altitude of 450 km. During this time, the spacecraft will slowly "walk-out" of the atmosphere by gradually raising its periapsis altitude to 143 km. Daily ABMs will be performed to maintain a guaranteed worst-case, two-day orbit lifetime. In other words, in the absence of ABMs due to unforeseen events that inhibit the ability of flight controllers to send commands, the spacecraft will always be at least two days from crashing into the surface.

Walk-out phase will also represent an extremely critical time with respect to the power situation on the spacecraft. During most of the aerobraking main phase, the spacecraft will never encounter regions in the orbit where Mars eclipses the Sun from the solar arrays. Consequently, the only battery discharge times will occur during the drag pass when the solar arrays will be oriented in an aerodynamic favorable orientation rather than in a power-collection configuration. In walk-out, the spacecraft will experience both long drag passes and eclipse zones on every orbit. The maintenance of a favorable energy balance situation on the spacecraft will be further complicated by the fact that the short two-hour orbits in walk-out will provide little time to recharge the batteries between successive drag passes.

5.3.5 Aerobrake Termination

Aerobraking will end with a termination burn (ABX) performed sometime during mid-January 1998. This burn will raise the orbit periapsis out of the atmosphere to an altitude of approximately 450 km. At this time, the spacecraft will be circling in a 400 x 450 km orbit with a period slightly under two hours. In addition, the descending node location will have regressed from its original MOI position at 5:45 p.m. with respect to the fictitious mean Sun to nearly 2:00 p.m.

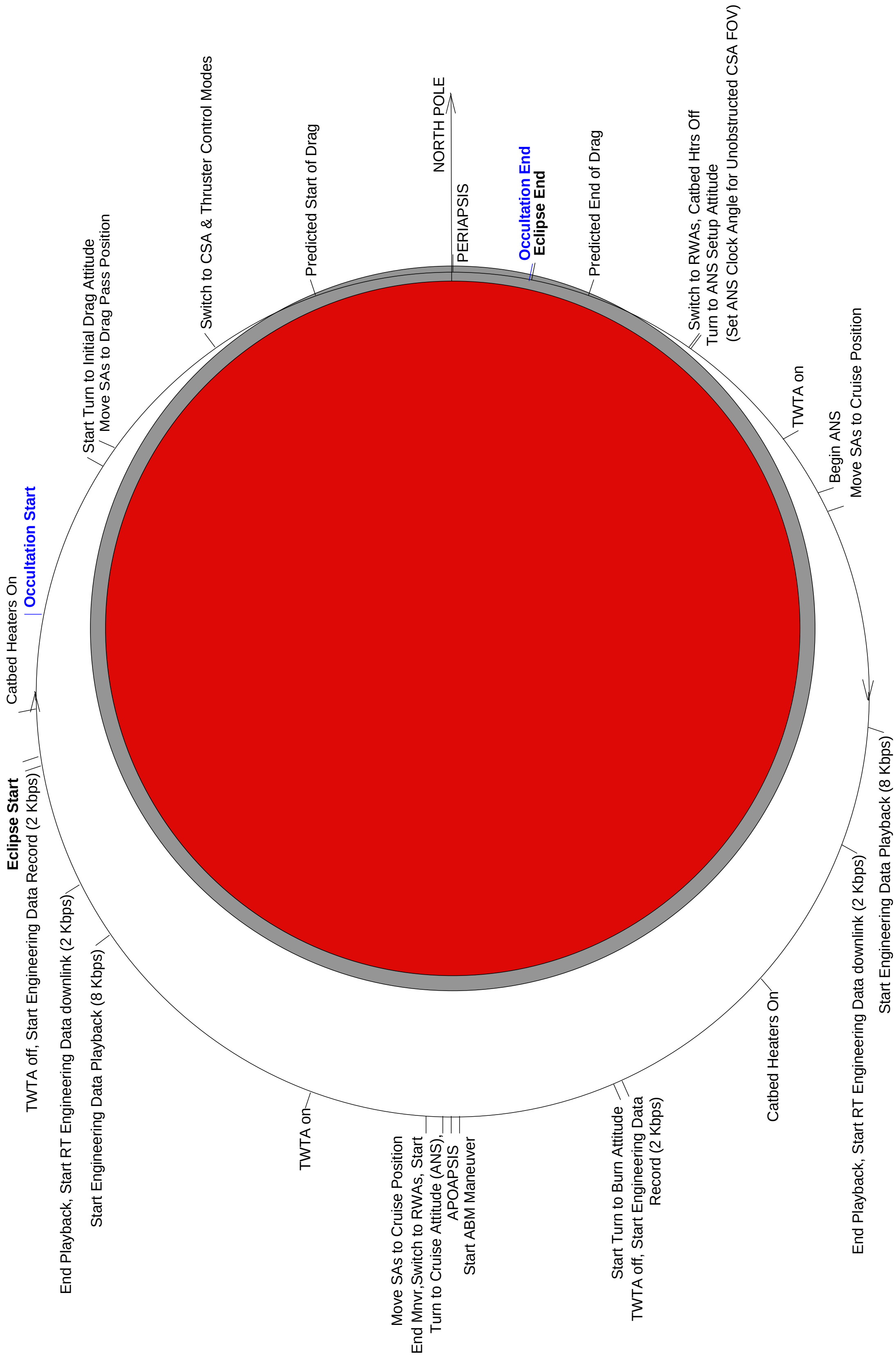
5.3.6 Spacecraft Drag-Pass Activities

Figure 5-5 shows a sample event profile plot for a typical aerobraking orbit. The events in the orbit will occur relative to the predicted time of periapsis passage and will be executed from an onboard, stored command sequence. Timing pads added to both sides of the periapsis passage will allow for navigation uncertainties in the periapsis crossing time. As the orbit period decreases, the drag-pass durations will increase. Consequently, the timing between events in the sequence will need to be updated on a periodic basis.

Throughout the aerobrake drag-pass sequence, high-rate engineering telemetry at 2 kbps will be recorded on one of the SSRs. This sequence will start with a 20-minute warm-up of the AACS thruster's catbed heaters. Then, the transmitter will turn off to conserve power. Approximately 7.5 minutes prior to the predicted start of the drag pass (including margin for navigational uncertainty), the spacecraft will turn to a "tail-first" attitude under reaction wheel control. The desired entry attitude will point the -Z axis along the velocity vector to keep the science instruments and solar cells pointed away from the incoming air flow.

Once the turn completes, the solar arrays will then gimbal to their drag-pass positions of 30^o above the Y-axis in the direction of +Z. The gimbaling to this configuration will be accomplished by rotating the inner (elevation) gimbals by 90^o from their ANS positions. The 30^o sweep-back angle will provide

Figure 5-5: ***Sample Event Profile of an Aerobraking Orbit***



for maximum aerodynamic stability with minimal drag reduction by moving the center-of-pressure aft of the center-of-gravity. This configuration will be aerodynamically stable because the aerodynamic torque will push the attitude back toward the aerodynamic null, a point where the velocity will align with the -Z axis. Additionally, this orientation will also place the solar arrays against their gimbal hard-stop positions to prevent back driving the gimbal motors.

At the predicted start of the pass (including margin for navigational uncertainty), attitude control authority will switch from reaction wheel to the AACS thrusters, and the wheels will be placed in TACH-HOLD mode in order to maintain their current spin speeds. The reason for this switch is that aerodynamic torques on the spacecraft during the drag pass will be larger than the reaction wheels can counteract. In order to minimize the number of thruster firings and the amount of AACS propellant consumed during aerobraking, attitude-control deadband regions will be opened up to $\pm 15^{o}$. This scheme will allow the spacecraft's attitude to oscillate within 15^{o} of the aerodynamic null without consuming any propellant. At the predicted time of periapsis, the spacecraft will perform a controlled momentum desaturation by setting the TACH-HOLD wheel speeds to zero.

During the drag pass, the spacecraft will attempt to keep the -Z axis along the velocity vector and the +X axis nadir pointed by using the CSA in an aerobraking attitude-control mode. This mode will utilize the onboard ephemeris to provide position knowledge about Mars. Unfortunately, buffeting of the spacecraft by atmospheric turbulence will dramatically increase the chances of star misidentification by the CSA. In order to prevent this potential misidentification, star processing by the CSA will be disabled during the drag pass, and the gyros in the IMU will propagate the spacecraft's inertial-attitude reference.

At the predicted end of the drag pass, the attitude-control deadbands will be tightened back to normal, and the reaction wheels will assume attitude-control authority from the AACS thrusters. Then, the spacecraft will slew back to an ANS orientation and point the +X axis at the Earth. For very small orbit periods of less than three hours in duration, there exists a concern that Mars may block part of the CSA's field of view upon return to ANS. This blockage may further extend the amount of time that the spacecraft's inertial reference remains in a state not updated by star scans from the CSA.

One option to mitigate the Mars-blockage situation involves using ISH mode to rotate the spacecraft by 180^{o} around its Y axis before returning to ANS. The net effect of the rotation will cause the +Z axis of the spacecraft to flip from pointing backward along the velocity vector to pointing forward. Because the CSA sits attached to one corner of the nadir panel on the +Z end, the "flip" will cause the CSA to point in an opposite direction. When the spacecraft returns to ANS from this flipped orientation, Mars will not block the CSA field of view.

The 180^{o} Y-axis slew to flip the +Z side of the spacecraft and reposition the CSA will result in an added bonus. As the +Z panel moves from pointing backward along the velocity to pointing forward, the panel will slew across the Martian surface because the velocity vector will lie approximately tangent to the surface. Consequently, the science instruments will be able to take data during this 10-minute slew because they sit on the +Z panel of the spacecraft. During the slew, the view geometry will vary from a horizon view at the beginning, to a direct nadir view in the middle, and to the opposite horizon view at the end.

After returning to an Earth-pointed ANS orientation following the completion of the post-drag-pass slew, the onboard sequence will activate the transmitter and the spacecraft will begin a period of real-time broadcast of engineering telemetry at the 2-kbps rate. This broadcast will allow the flight team to assess the current status of the spacecraft and its systems. A playback of the engineering telemetry recorded on the SSRs during the drag pass will follow the realtime transmission period.

5.3.7 Corridor Control Maneuvers

As described earlier, the periapsis altitude during aerobraking must lie within a well-defined corridor low enough to produce enough drag to reduce the orbit within the time constraints to reach the 2:00 p.m. node, yet high enough to avoid spacecraft heating and maximum dynamic pressure limits. As often as once per day, the spacecraft will execute maneuvers at apoapsis to maintain the periapsis altitude within this corridor. These maneuvers are called ABMs.

The ABM command sequence will be pre-loaded in the spacecraft's memory and initiated by real-time command as dictated by the flight team. As part of the ABM command process, the flight team will upload key parameters to the onboard command script. Some of these parameters include the burn time (almost always at apoapsis), burn direction, and burn magnitude.

In order to expedite the ABM command process, the flight team will select the maneuver parameters from a predefined menu of choices. This menu will contain only two burn directions and three burn magnitudes. The burn direction will be either UP for a periapsis raise maneuver, or DOWN for a periapsis lowering maneuver. The list of burn magnitudes consists of HALF-SIZE, FULL-SIZE, and DOUBLE-SIZE.

Typically, the quantitative values that correspond to the parameter choices in the ABM maneuver menu will be valid for ABMs on several consecutive orbits. The flight team will update these values periodically throughout aerobraking. Updates will occur with increasing frequency toward the end of aerobraking due to the increasing pace of variation in the orbit geometry as main phase ends and walk-out begins.

5.3.8 Science Activities

The science instruments will collect a limited amount of data during the early and middle portions of aerobraking. Data collection will occur during the 10-minute time period after the aerobrake drag pass as the spacecraft slews its +Z panel across Mars from one horizon to the opposite horizon. This post-periapsis slew will represent both a unique opportunity for the scientists to gather high-resolution data due to the low drag-pass altitude (about 110 km as opposed to 378 km during mapping), and for the flight team to receive key information to support aerobraking operations. Due to spacecraft power constraints, the instruments will be powered off when the orbit period has shrunk to less than six hours.

Specifically, the flight team will use the science data generated during aerobraking to help in the understanding of the density and the orbit-to-orbit variability of the Martian atmosphere. For example, images generated by the MOC may provide insight into the detection of local and global dust storms. An early warning into the formation of these storms will be crucial because the storms tend to heat the atmosphere and increase its density. If the density of the atmosphere suddenly increases, then the flight team may need to immediately raise the altitude of periapsis to compensate.

The TES will also perform atmospheric scans during the post-drag slew. Additionally, because TES will sit on the spacecraft in a position pointing backward along the velocity vector during the drag-pass, the instrument will be able to directly scan the regions of the atmosphere that the spacecraft has just recently flown through. These scans will allow the TES team to supply quantitative information to the navigation team in the form of temperature profiles of the lower atmosphere with a one kilometer resolution. An understanding of the temperature profile of the lower atmosphere will yield clues into the density and variability of the upper atmosphere.

In addition to the MOC and TES, the MOLA and MAG will also take measurements during the post-drag-pass slew. Although this data will be useful from a scientific perspective, it will not support aerobraking operations because no insight can be gained into the density of the Martian atmosphere from these two instruments.

5.4 Transfer to Mapping

The transfer-to-mapping phase will be the final period of the orbit insertion phase. Three critical events will occur in this time period. The first is the acquisition of the mapping orbit. Second, a gravity calibration will be performed to update the gravity field model. Finally, the spacecraft will be deployed into its mapping configuration and the instruments will activate in preparation for the start of mapping.

5.4.1 Mapping Orbit Acquisition

The mapping orbit for MGS is a low-altitude, near-circular, near-polar orbit which is Sun-synchronous with the dayside equatorial crossing at 2:00 p.m. mean solar time, all of which have nearly been achieved upon completion of the aerobraking phase. In order to meet altitude variation requirements in mapping for the instruments and the horizon sensor (MHSA), the mapping orbit must also be a frozen orbit with a stationary periapsis location around the South Pole. Upon completion of ABX, the periapsis latitude is 6° S for the open of the launch period and is 42° N for the close of the launch period. Propulsively moving periapsis down to the South Pole is beyond the capability of the propulsion system. The effect of the gravity field, however, is to move periapsis towards the South Pole at approximately 3.25 °/day. Thus periapsis reaches the South Pole in 26 days for the open of launch period case and 37 days for the close of launch period case. Once periapsis is in position, the Transfer to Mapping Orbit (TMO) maneuver is executed with the main engine to acquire the mapping orbit.

5.4.2 Gravity Calibration

In order to improve prediction and reconstruction of the spacecraft ephemeris early in the mapping phase, Navigation desires a gravity calibration period in order to update the gravity field model. Ideally the gravity calibration would be performed over a 7 sol or 7.2 day (1 sol equals a Martian day) period after acquisition of the final mapping orbit. The mapping orbit provides a ground track pattern that nearly repeats in 88 orbits or seven Martian days, providing uniform coverage of the planet with a maximum ground track spacing at the equator of about 240 km. This provides good resolution for the gravity field solutions. However, because of the extended orbit insertion timeline up to this point, in the interest of expediting the start of mapping, the gravity calibration and subsequent model update is instead performed during the extended orbit drift period between ABX and TMO.

During the gravity calibration period, the DSN will collect a continuous Doppler data set, except during occultations. The spacecraft will remain in the cruise configuration during the gravity calibration period, rotating slowly about the Earth-pointed +X axis. This configuration will minimize HGA disturbances that could degrade the Doppler measurements.

Twelve days after completion of TMO, the first Orbit Trim Maneuver (OTM1) is performed to freeze the mapping orbit based on the updated gravity model. As a result of the expected 3σ ΔV magnitude for this maneuver, OTM1 is executed with the main engine. Due to contamination of the deployed HGA during a main engine burn, the HGA cannot be deployed to its mapping position until completion of OTM1. In other words, once the HGA has been deployed the main engine is no longer utilized for maneuvers.

5.4.3 Mapping Configuration

Upon completion of OTM1, a period of ten days is reserved for deployment of the spacecraft into the mapping configuration, powering on the instruments and to perform a mapping rehearsal. These events must be completed before declaring the start of the mapping phase.

On the first day of the mapping configuration period, the oxidizer side of the propulsion system is isolated by closing the normally open pyro valve. The next day the HGA will be deployed to its final mapping position. The following day, an HGA calibration is performed to determine the exact position of the HGA boom and to update, if necessary, the HGA gimbal zero-reference point offset parameters in the flight software.

On the fourth day of the mapping configuration period, the Mars Horizon Sensor Assembly (MHSA) is powered on for 48 hours prior to use in order to characterize its performance before initiating mapping control. Flight software parameter uploads required to configure the spacecraft for mapping operations are also uplinked to the spacecraft on this day. Finally on day five, the spacecraft is commanded to begin mapping nadir pointing. The spacecraft is monitored throughout the next day to characterize its operations in the mapping configuration.

On day seven, the instruments are powered on and their memories uploaded as required. The PDS is then commanded to the LRC mode (4 ksps data rate for realtime and/or recording) and continual recording initiated on the recorders. Over the next two days a mapping rehearsal is performed to verify the mapping data return strategy. Specifically the use of autonomous eclipse detection algorithm (AEM) to sequence the data return is verified. A brief description of AEM follows in Section 5.3.4 with regards to powering off the transmitter during eclipse periods during the post aerobraking period before the start of mapping. Detailed description of AEM and its use for sequencing the mapping data return is provided in Section 6. Upon completion of the mapping rehearsal, the spacecraft will be declared ready for mapping and the mapping phase initiated.

5.4.4 Spacecraft Activities

In the mapping orbit, the power subsystem cannot support the transmitter high power beam on during the approximately 40 minute solar eclipse period each orbit. During aerobraking, the transmitter was cycled on and off as part of the drag pass sequence of events. Upon completion of aerobraking, however, Autonomous Eclipse Management (AEM) is enabled to manage the cycling of the TWTA beam high power during the solar eclipse period of each orbit for the remainder of the mission. AEM is an autonomous eclipse detection algorithm which detects when the spacecraft enters and exits eclipse and initiates ground specified command scripts, in this case to power the transmitter off and on, respectively. Use of AEM during the transfer to mapping sub-phase allows characterization of this capability for use during mapping operations when AEM is additionally utilized to sequence the science data return events.

After reaching the mapping orbit and completing the gravity calibration period, the high-gain antenna (HGA) will be fully deployed and verified. The HGA deploy sequence begins with a 20 minute warm up of the thruster catbed heaters. Next communications are switched from the HGA to the primary LGA transmit antenna mounted on the HGA, to ensure downlink throughout most of the deployment and in the event the boom does not deploy properly. There is insufficient link margin to transmit the high rate engineering telemetry over the LGA, so the EDF is commanded to emergency mode to provide 10 bps downlink. A couple of minutes prior to the deployment, the spacecraft is commanded to the “deploy/despin” attitude control mode and actuator control is switched to the thrusters. In this mode, the four reaction wheels are held in tach hold at or above 200 rpm to protect them from possible shock damage when the retention and release devices are fired. The spacecraft controls to a desired attitude throughout the deployment. The HGA is deployed by simultaneously actuating three retention and release devices. The boom/antenna assembly rotates roughly 150^{o} and latches into place within 10 minutes. Actuator control is switched back to reaction wheels and the normal cruise “array-normal-spin” mode is reacquired. The HGA is then rotated around to align the boresight back to the Earth. This is accomplished by rotating the outboard or azimuth gimbal roughly 180^{o} followed by rotating the inboard or elevation gimbal about 30^{o}. Based on the signal strength an approximate measure of any boom displacement can be made and a determination to re-enable communications over the HGA. An HGA calibration is planned in order to determine the exact position of the boom. Upon determination of the actual boom position, the gimbals' zero reference points are updated as required.

After verification of the HGA deployment, the IMU is commanded to the "low rate" mode to meet required mapping pointing accuracy. Additionally, the MHSA is turned on 48 hours prior to initiation of mapping nadir pointing control, in order to verify its health and characterize its operation. Various flight software parameters updates for mapping operations are subsequently uploaded to the spacecraft.

At this point the spacecraft is now ready to acquire the mapping nadir pointing attitude. In order to do this, the spacecraft is first commanded to the "inertial slew / hold" attitude control mode to slew the spacecraft such that the +Z axis is pointed at Mars. Using this mode provides autonomous payload Sun avoidance protection during Mars acquisition, which is a capability not available in the mapping attitude control modes. Attitude control is then switched to "CSA / Mapping" mode to point and maintain the spacecraft +Z axis along the velocity vector. Once the MHSA has acquired "Mars Lock" in which all four quadrants are viewing Mars, attitude control is autonomously switched to "Primary" mode. In primary mode, roll and pitch control are maintained using the MHSA, while yaw is controlled using the IMU as a gyrocompass. Additionally, using the planetary ephemeris, autonomous HGA Earth tracking and solar array Sun tracking are enabled. After a period of on-orbit characterization, the spacecraft is declared ready for mapping.

5.4.5 Science Activities

The only science activities planned during the transfer-to-mapping phase are MAG data collection throughout most of the phase and the final MOC focus check, scheduled for 31 January 1998. The MAG / ER and the PDS are powered back on after ABX and left on until the start of the spacecraft mapping deployment. The MOC focus check is the last science activity and represents the low temperature regime in establishing the focus heater control authority.

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6. Mapping Phase

Mapping phase represents the period of concentrated return of science data from the mapping orbit. This phase will start on 15 March 1998 and last until 31 January 2000, a time period of one Martian year (687 Earth days). These dates will remain fixed and are independent of the actual day of lift-off within the launch period. During this phase, the spacecraft will keep its science instruments (+Z panel of the spacecraft) nadir pointed to enable data recording on a continuous basis. On a daily basis, the spacecraft will transmit its recorded data back during Earth during a single 10-hour Deep Space Network (DSN) tracking pass. An articulating high-gain antenna (HGA) on the spacecraft will allow data recording to proceed while downlink to Earth is in progress. A timeline of mapping activities is shown in Figure 6-1

6.1 Mapping Orbit Design

Designing the proper mapping orbit to fulfill the science objectives, and to comply with Mars planetary protection requirements required taking many constraints into account. Section 3 of the MGS Investigation Description and Science Requirements Document (ID-SRD, JPL D-12487 or MGS 542-300) contains detailed information regarding the trajectory related science requirements. In general, the ID-SRD requirements are satisfied by a low-altitude, near-circular, near-polar, Sun-synchronous orbit with a "short" repeat cycle.

The baseline mapping orbit chosen for the mission will utilize a descending node orientation of 2:00 p.m., index altitude of 378 kilometers (semi-major axis of 3774.998 km), near-circular eccentricity of 0.00953, equatorial inclination of 93.011^{o}, and argument of periapsis of -90^{o}. This combination will result in a "frozen orbit" with a nodal period of approximately 117.64 minutes. In other words, the frozen orbit condition will cause the argument of periapsis to always remain close to -90^{o}, a location near the Martian South Pole. These orbital elements were derived using the Mars 50c 50 x 50 gravity field model. Prior to

Figure 6-1: ***Timeline for Mapping Phase***

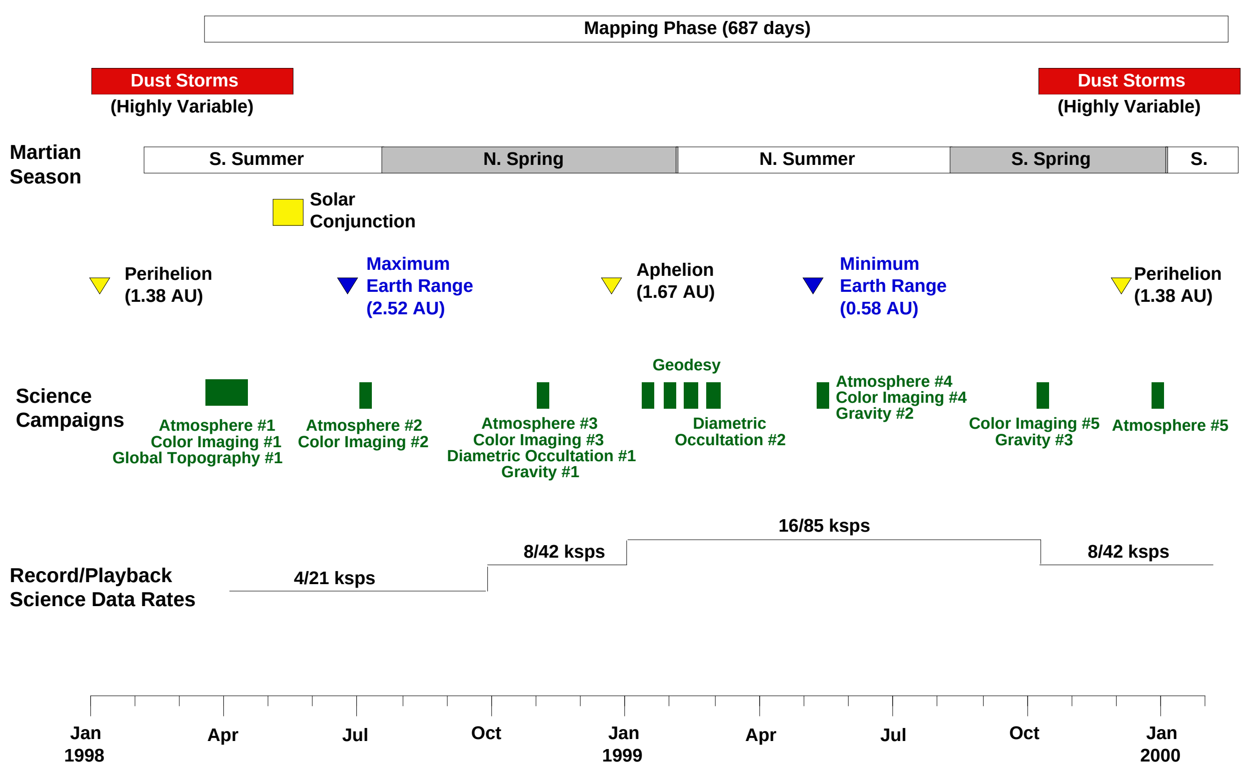
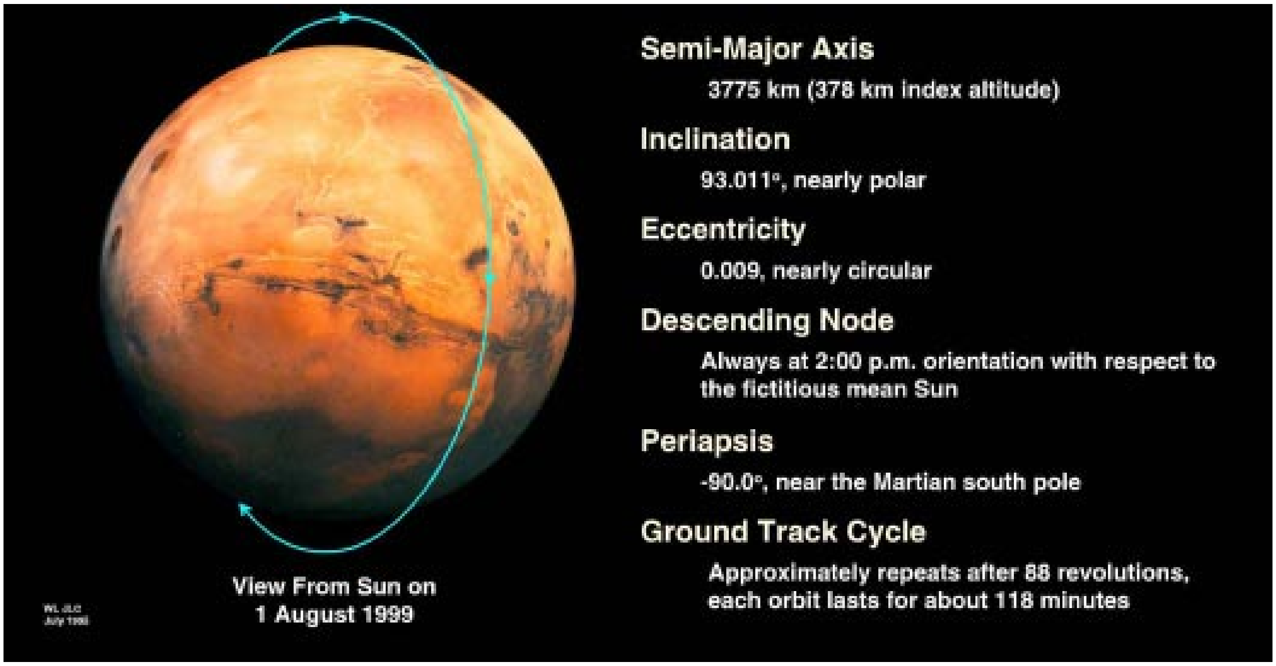


Figure 6-2: ***View of the Mapping Orbit***



the start of mapping, this gravity field model will be refined using radiometric data collected during gravity calibration activities. Subsequently, the mapping orbit will be adjusted to be consistent with the new field model.

6.1.1 Basis for Mapping Orbit Selection

In general, the Mars Orbiter Camera (MOC) team desired an orbit with a node time late in the afternoon for longer shadows, while the Thermal Emission Spectrometer (TES) preferred an orbit closer to 1:00 p.m. to maximize the signal-to-noise ratio in their data. Also, the Magnetometer (MAG) team asked for an orbit fixed in local time to minimize the impact of large scale ionospheric currents on their observations. The 2:00 p.m. Sun synchronous, descending node orbit orientation represents a compromise between MOC and TES that was also acceptable to the MAG.

Constraints on mean altitude and inclination were primarily driven by the MOC and the Mars Orbiter Laser Altimeter (MOLA). The MOC requirements called for an average mapping orbit altitude less than 540 km to guarantee image resolutions better than two meters per pixel, while the MOLA needed an orbit inclination within 3^o of polar with altitude variations less than 105 km. Use of a 378-km index altitude for mapping operations satisfies the MOC low-altitude requirement. Index altitude is a measure of the mean semi-major axis, and reflects the difference between the semi-major axis length and the equatorial radius of Mars (3397.2 km). This parameter is related to, but not the same as the average height of the spacecraft above the surface of Mars. The 378-km number was chosen in large part for its desirable properties with respect to ground track repeat cycles and spacing.

Setting the semi-major axis to a specific value constrained the valid choices for eccentricity and inclination in terms of the spacecraft maintaining a Sun-synchronous orientation. Although many possible solutions exist, the spacecraft will fly in a 93.011^o orbit with an eccentricity of 0.009. These parameters satisfy the MOLA constraint for a near polar, near circular orbit with minimal altitude variations.

One of the biggest advantages for the specific choice of semi-major axis, inclination, and eccentricity is that the mapping orbit can subsequently be “frozen” by setting the argument of periapsis to -90^o, almost directly over the Martian south pole. This orbital configuration will keep the mean values for eccentricity and argument of periapsis fixed. Essentially, the secular motion of periapsis due to J_2 is canceled by the long period variation due to the J_3 component of the gravity field.

For the MGS frozen orbit, the range of areocentric altitudes over a single orbit will always remain between 345 km (south pole) to 417 km (north pole), satisfying the MOLA constraint for less than 105 km altitude variation. Without the frozen orbit, the variations in eccentricity and periapsis location will cause the altitude range to exceed the horizon sensor's (MHSA) operational limits of 335-km minimum to 455-km maximum. Such an excursion would also violate the MOLA's 105-km altitude variation constraint.

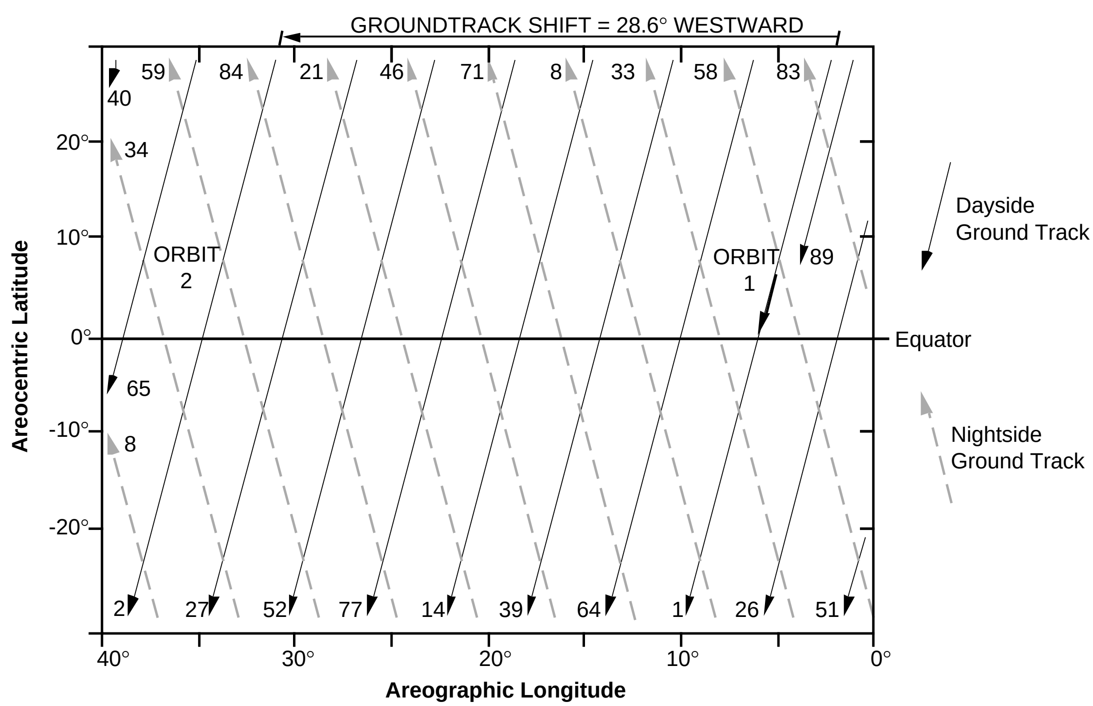
A frozen orbit with a +90^o argument of periapsis (over the Martian north pole) is also mathematically possible. However, this option was not chosen because it would force the eccentricity to take on a higher value than 0.009. Consequently, the orbit would appear less circular in shape and the altitude variations would correspondingly increase.

6.1.2 Ground Track Repeat Cycle

A 378-km index mapping altitude yields an 88 revolution near-repeat cycle lasting about seven sols (Martian day). In other words, the pattern of ground tracks under the spacecraft will nearly repeat every 88 orbits. Technically, the mapping orbit was designed for a Q value of 6917/550. This parameter denotes the number of revolutions completed by the spacecraft in one Martian day. Therefore, if the mapping orbit is perfectly maintained, the ground track pattern will exactly repeat after a "super cycle" of 6917 orbits in exactly 550 sols. However, maintaining this pattern will not be possible due to navigational uncertainties in controlling and predicting the orbit. Atmospheric drag represents the most difficult orbit perturbation to measure.

In practice, the Q value of the mapping orbit will be roughly 88/7. Since 88/7 represents only an approximate fractional reduction of 6917/550 (12.57143 vs. 12.57636), the ground track pattern will not repeat exactly on every cycle. Instead, after every 88-orbit repeat cycle, the ground track pattern for that cycle will appear offset slightly to the east of the pattern laid down by the previous cycle. This offset

Figure 6-3: ***Mapping Orbit Ground Track Pattern for Seven-Sol Repeat Cycle***



amounts to 58.6 km when measured at the equator. For example, the equatorial location of ground track number 89 will lie 58.6 km to the east of track number 1. This 58.6-km differential is called the "orbit-walk."

The 58.6-km orbit-walk at the equator does not represent the spacing between ground tracks on two successive orbits. During each mapping orbit lasting 117.65 minutes, Mars turns under the orbit from west to east at a rate of approximately 0.24^{o} per minute. This rate of planetary rotation will cause the spacecraft to fly over locations about 28.62^{o} to the west of the locations flown over on the previous orbit. At the equator, 28.62^{o} translates to a ground track spacing of 1,697 km between two successive orbits. Ground tracks laid down by later orbits within the 88-orbit repeat cycle will gradually fill the 1,697-km gap between two consecutive orbits. Figure 6-3 illustrates the concepts discussed in this paragraph.

6.1.3 Sun and Earth Obscuration Geometry

The percentage of time during the mapping orbit when Mars obscures the spacecraft's view of either the Earth or Sun will directly impact spacecraft sequencing and the data gathering and return strategy. Obscuration durations depend directly on the beta angle to the target in question. Target betas measure the angle between the plane of the orbit and a vector to the target, either the Sun or Earth. In other words, the beta angle provides a measure of the angular distance between the target and mapping orbit plane. Specifically, the beta angle is 90^{o} minus the angle between the vector to the target and the angular momentum vector of the orbit.

If an orbit appears "face-on" to the target (view from the target perpendicular to the plane of the orbit), the beta angle will measure 90^{o} and no obscuration of the target occurs as seen from the spacecraft. In other words, the spacecraft will spend no time "behind" Mars relative to the target. Beta values for no obscuration can fall lower than 90^{o} with a high enough orbital altitude. If an orbit appears "edge-on" to the target (view from the target parallel to the plane of the orbit), the beta angle will measure 0^{o} and the target will be obscured for a maximum amount of time. As a general rule of thumb, longer obscurations will result when the target is closer in position to the mapping orbit plane than when farther away. Figure 6-4 illustrates the range of angular separations between the Earth, Sun and the MGS mapping orbit.

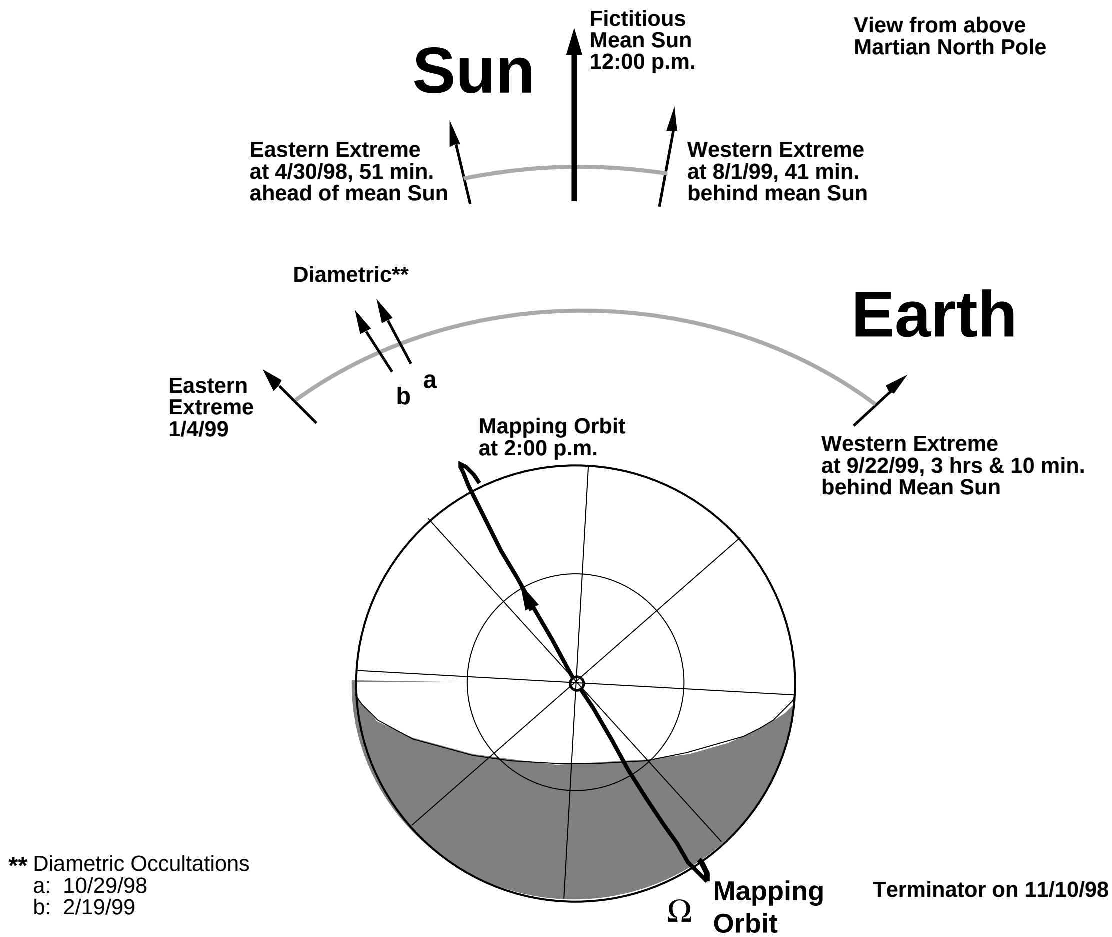
Because the mapping orbit lies fixed at a 2:00 p.m. orientation relative to the fictitious mean Sun at Mars, a 90^{o} beta angle means that the target lies at either 8:00 a.m. (90^{o} west of the orbit) or 8:00 p.m. (90^{o} east of the orbit). On the other hand, a 0^{o} beta angle indicates that the target's position in the sky also lies at 2:00 p.m. relative to the mean Sun at Mars. In general, this type of analogy requires the use of caution because 15^{o} of beta angle does not always translate into one hour of mean solar time differential. The reason is that while the fictitious mean Sun always lies in the Martian equatorial plane, the target may not lie there.

6.1.4 Earth Occultations

During the mapping phase, the position of the Earth will change drastically with respect to the mapping orbit plane (see Figure 6-4). Earth occultation durations will last for roughly 40 minutes out of every 117.64 minute orbit at the start of mapping in March 1998. At this time, the Earth's position will be at about 12:30 p.m. relative to the fictitious mean Sun at Mars, roughly 15^{o} to the west of the mapping orbit plane.

Figure 6-5 plots the Earth occultation duration during mapping. As time progresses forward from the start of mapping, the Earth will gradually move eastward toward the mapping orbit plane, and occultation durations on every orbit will increase as the beta angle drops to 0^{o} on 29 October 1998. On that day, the orbit plane will appear "edge-on" to the Earth, and occultation durations will reach a maximum of 41.3 minutes out of every revolution.

Figure 6-4: ***Earth and Sun Positions Relative to Mapping Orbit***



About three and one half months later on 19 February 1999, the Earth's position as seen from Mars will pass through the orbit plane yet another time, but this time on its way back west from the east. Again, occultation durations will reach a maximum due to the "edge-on" orbital geometry. Then, the occultations will slowly decrease in duration as the Earth continues to move westward, away from the mapping orbit plane.

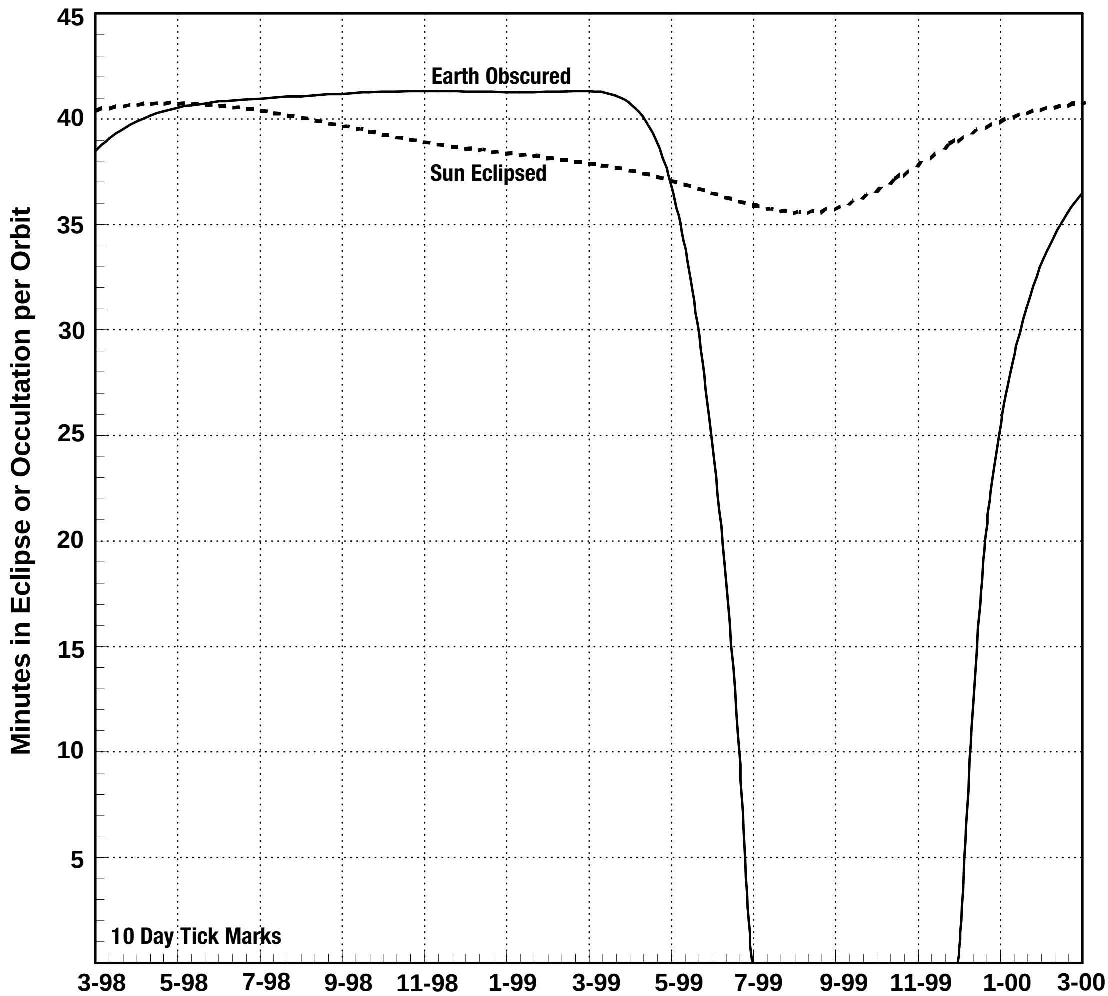
Four months later on 19 June 1999, the Earth will reach a position nearly 70^o to the west of the mapping orbit, roughly 9:05 a.m. relative to the fictitious mean Sun. This date is significant because despite not yet reaching a "face-on" orientation to Earth, the orbit lies at sufficiently high altitude that the spacecraft will no longer pass behind Mars relative to Earth. In other words, Earth occultations will no longer occur during any part of the mapping orbit, and a clear line of sight will always exist between the Earth and spacecraft.

The Earth's position as seen from Mars will continue to move westward until 22 September 1999 when it reaches an extreme of 8:50 a.m. relative to the fictitious mean Sun. Then, the Earth will begin to move back toward the east. Occultations will begin to occur again on 26 November 1999 and will increase in duration for the remainder of the mapping phase of the mission.

6.1.5 Solar Eclipses

Despite using a Sun-synchronous mapping orbit, the position between the spacecraft's orbit plane and the Sun will oscillate as a function of time. Consequently, both the solar beta angle and the amount of

Figure 6-5: ***Earth Occultation and Solar Eclipse Durations During Mapping***



time during an orbit that Mars eclipses the Sun from the spacecraft will also vary (see Figure 6-5). The reason for this variation is that the 2:00 p.m. Sun-synchronous orientation is defined with respect to the fictitious mean Sun, and Mars' obliquity and its orbital eccentricity around the Sun cause the position of the true Sun to oscillate about the mean Sun. This position will vary depending on the Martian season and the location of Mars in its orbit around the Sun. During the mapping phase, the true Sun's position will vary from a maximum of roughly 51 minutes ahead (east of) to a minimum of about 41 minutes behind (west of) the fictitious mean Sun. Because the mapping orbit plane lies two hours ahead of the fictitious mean Sun, the true Sun will always be to the west during mapping despite the positional variations.

The longest amount of time during one orbit that Mars eclipses the Sun occurs when the true Sun reaches its most eastern position (closest to the mapping orbit) in late April, early May 1998. At that time, eclipses will last for 40.7 minutes out of every 117.64 minute revolution. On the other hand, the shortest eclipse durations occur when the true Sun arrives at its most western position (farthest away from the mapping orbit) in late July, early August 1999. Then, eclipses will only last for about 35.5 minutes every orbit.

6.1.6 Mapping Orbit Numerical Parameters

Table 6-1 lists the predicted orbital elements of the mapping orbit, both mean values and osculating at periapsis. All values in the table are referenced to the Mars centered, Mars mean equator and IAU

Table 6-1: ***Mapping Orbital Elements (Pre-Launch Prediction)***

Orbital Element	Mean Value	Osculating at Periapsis
Semi-Major Axis	3774.998 km	3765.939690
Eccentricity	0.00953	0.006338
Inclination	93.011^o	93.014163^o
RA of the Ascending Node	varies (see text)	varies (see text)
Argument of Periapsis	270^o	270^o

vector of epoch coordinate system, reference epoch 15 January 1998 at 1:00 ephemeris time (ET) for both the orbit and IAU vector. These elements were derived using the Mars 50c 50 x 50 gravity field model. Although the elements listed in the table are extremely accurate with respect to the current knowledge of Mars' gravity field, the exact values are also highly dependent on the initial conditions. Consequently, the orbit will be adjusted prior to the start of mapping using radiometric navigation data collected during the post-aerobraking gravity calibration period.

During mapping, the orbital elements will be controlled to keep the spacecraft in a Sun-synchronous orientation with a descending node crossing within three minutes of 2:00 p.m. ($\pm 12^o$ from the normal position). As a result, the right ascension of the ascending node (Ω) will advance with time at a rate that matches the mean motion of Mars around the Sun. The expression below provides a way to calculate Ω at any arbitrary time during the mission.

$$\Omega = -98.382323^o + 0.524040(t - t_o)^o$$

In this expression, t is number of Earth days, t_o is 10 September 1997 at 0:00 ET, and Ω is in degrees referenced to the Mars mean equator, IAU vector coordinate system. Keep in mind that the 2:00 p.m. orientation is with respect to the position of the fictitious mean Sun, not the location of the true Sun. With respect to the true Sun, the descending node time of the orbit will vary from about 1:08 p.m. to 2:41 p.m. depending on the Martian season and Mars' exact location in its orbit around the Sun.

Based on the pre-launch mapping orbital element predictions, a trajectory was integrated forward for two years to determine the important events from an Earth and Sun view geometry and Martian season perspective. Table 6-2 lists these events.

6.2 Spacecraft Configuration

While in mapping, the two solar arrays and the HGA (deployed on a two meter boom) will be gimbaled in two degrees of freedom to track the Sun and Earth while the spacecraft moves around the 117.64 minute orbit. In the primary attitude control mode, data from the horizon sensors (MHSA) and the IMU will collectively be used to point the science instruments on the +Z spacecraft panel in the nadir direction, and to point the +X axis in the direction of orbital motion. A backup control mode will also be available to point the spacecraft using data from the star sensor (CSA) and an ephemeris prediction loaded from the ground. Such a scheme will probably require updating the ephemeris on a weekly basis in order to maintain the 10-mrad pointing requirement.

6.2.1 Orbit-Trim Maneuvers

A set of three reaction wheels will point the spacecraft under normal circumstances. However, due to atmospheric drag, gravity gradient, and solar pressure torques, the momentum in the wheels will build up and must be periodically unloaded by autonomous thruster firings of the attitude control jets. Most of the unloading burns will occur near periapsis (around the south pole). For typical orbit-trim maneuvers (OTM) with a velocity increment in the +X or -X direction, the spacecraft will turn under reaction wheel

Table 6-2: ***Important Geometrical Events During Mapping***

Date	Event	Comments
06-Feb-98	Winter Solstice	Start of southern summer
30-Apr-98	Solar Beta Angle Maximum	Beta angle at -15.909^o, solar eclipse maximum of 40.72 minutes
04-May-98	Begin No Commanding Period	Start solar conjunction period, Sun-Earth-Mars angle < 2^o
12-May-98	Conjunction	Sun-Earth-Mars angle reaches minimum of 0.04^o
21-May-98	End No Commanding Period	End solar conjunction period, Sun-Earth-Mars angle > 2^o
22-Jun-98	Mars to Earth Range Maximum	Distance at 2.518 AU
15-Jul-98	Vernal Equinox	Start of northern spring
08-Jul-98	Northern Declination Maximum	Mars declination at 24.04^o as seen from Earth
29-Oct-98	Diametric Occultation (Edge-On-Orbit)	Earth beta angle at 0.0^o, Earth occultation maximum of 41.34 minutes
17-Dec-98	Mars Aphelion	Solar distance at 1.666 AU
04-Jan-99	Earth Beta Angle Maximum	Beta angle at 4.891^o
29-Jan-99	Summer Solstice	Start of northern summer
19-Feb-99	Diametric Occultation (Edge-On-Orbit)	Earth beta angle at 0.0^o, Earth occultation maximum of 41.34 minutes
24-Apr-99	Opposition	Sun-Earth-Mars angle at 178.62^o
02-May-99	Mars to Earth Range Minimum	Distance at 0.578 AU
19-Jun-99	Begin No Earth Occultations Period	Earth beta angle at -64.4^o
01-Aug-99	Autumn Equinox	Start of southern spring
22-Sep-99	Earth Beta Angle Minimum	Earth beta angle at -77.656^o
15-Oct-99	Southern Declination Maximum	Mars declination at -25.21^o as seen from Earth
25-Nov-99	Perihelion	Solar distance at 1.382 AU
26-Nov-99	End No Earth Occultations Period	Earth beta angle at -63.2^o
25-Dec-99	Winter Solstice	Start of southern summer

control to point the desired attitude control thruster in the proper burn direction. A typical OTM will last for several seconds.

6.2.2 Effect of HGA Gimbal Limit on Radio Science

A set of two gimbals, called the inner and outer, will allow the spacecraft to point its HGA directly at the Earth. These gimbals work on the theory that the vector to the Earth can be decomposed into an “out of the mapping orbit plane” component and an “in the mapping orbit plane” component. One of the gimbals, called the outer gimbal, will compensate for the out-of-plane component by rotating the antenna through an angle equal to the angular difference between the vector to the Earth and the plane of the mapping orbit. This angle is related to the Earth beta angle, changes slowly over the course of the mission, and can be considered constant for any given day during the mapping phase.

In contrast to the out-of-plane angle, the in-plane component can vary from -180^o to +180^o over the course of a single orbit. Due to size constraints on the inner gimbals, they will only be able to rotate through a range of -155^o to +155^o. This limitation does not pose a problem for the return of recorded data from the mapping orbit because the extreme range of in-plane angles will always occur as the spacecraft passes over the “far side” of Mars with respect to Earth.

However, the inner gimbal limit will affect the return of radio science occultation measurements. This experiment will gather data about the Martian atmosphere by observing the spacecraft’s radio signal passing through the atmosphere as the spacecraft enters and exits the region of Earth occultation. During part of the mission’s mapping phase, the in-plane angle will exceed + 155^o at the time of Earth occultation entry or exit.

Earth occultations take place on 528 out of the 687 days of the mapping mission with the gap occurring between 19 June 1999 and 25 November 1999, inclusive. The inner gimbal limit will reduce the total days of occultation experiments from 528 to 495, an amount equal to 6% of the total possible radio science days. All of these lost days will occur in June and November 1999 near the days when the orbit geometry causes Earth occultations to vanish and reappear, respectively. Radio science can be recovered on these days by moving the spacecraft off of its normal nadir point attitude. However, such a solution will impact the other instruments' ability to gather data for several minutes during the occultation ingress and egress time periods.

6.3 Data Collection Strategy

Two primary modes exist for collecting science data. Most of the time, data gathering will involve continuously recording the data on the solid state recorders and then playing it back to Earth during the daily Deep Space Network (DSN) tracking pass. The other mode will return realtime, high-rate telemetry at the 40 or 80 ksps rate during additional tracking passes scheduled every third day. In addition, collection of radiometric (Doppler and range) data will occur whenever the spacecraft downlinks telemetry to Earth.

As described in Section 2, the S&E-1 and S&E-2 data streams used for science data collection will be Reed-Solomon encoded, resulting in 250 encoded symbol bits for every 218 raw data bits collected by the Payload Data Subsystem (PDS) from the science instruments. Output streams will leave the PDS already Reed-Solomon encoded. Further detail regarding the bit allocations within the data stream can be found in Appendix A.

6.3.1 Data Collection Strategy for Recorded Data (S&E1)

The basic strategy for collecting science during the mapping phase will involve recording both science and engineering data onto one solid state recorder (SSR) on a continuous basis, and then playing back 24 hours worth of data during a 10-hour DSN track pass normally scheduled once per Earth day. Scientific data collection will continue during the playback period to avoid missing portions of the planet that the spacecraft travels over during the playback. This mode is called S&E1.

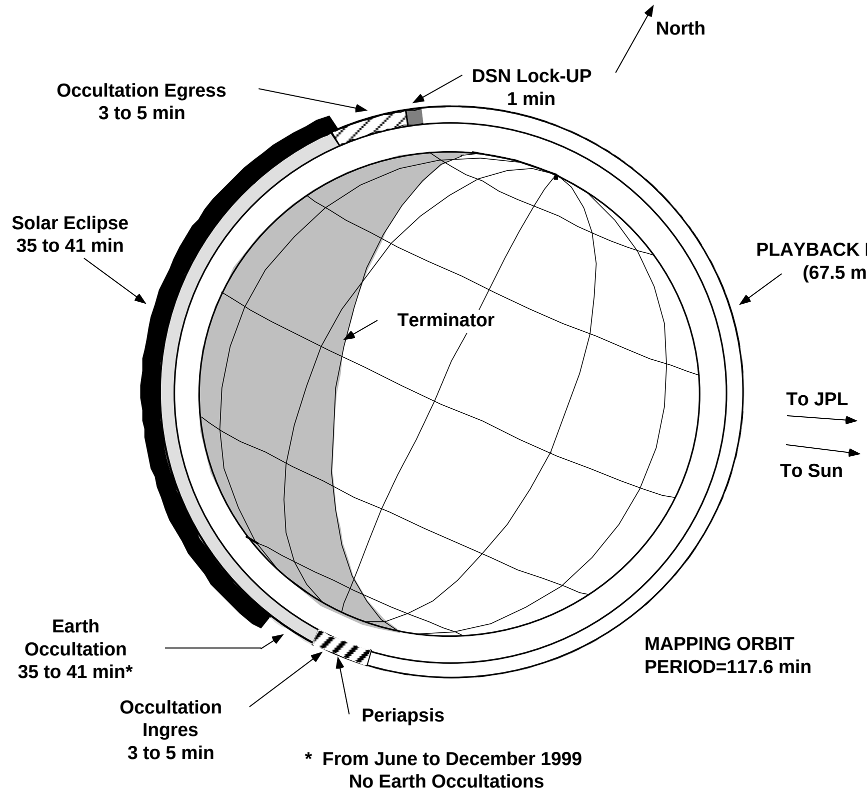
Playback Timing

During the standard, daily 10-hour DSN tracking pass for playback of recorded data (S&E1 mode), the spacecraft will circle Mars every 117.64 minutes and complete about five orbits. For most of the mapping phase, the spacecraft will enter zones of Earth occultation as it passes behind Mars relative to the direct line of sight to Earth (see Figure 6-6). The time that the spacecraft spends on the "back side of Mars" in the Earth occultation zone will typically eliminate 40 minutes out of every orbit from use for playback of the recorded science and engineering data.

Some additional time when the spacecraft exits occultation will be used for the atmospheric occultation experiment. This process works by transmitting to the Earth through the Martian atmosphere for a range of atmospheric altitudes from zero to 200 kilometers. Because telemetry modulation will be turned off for this experiment to provide maximum signal strength, the duration of this experiment will subtract from the available playback time. Normally, the exit experiment will typically require about five minutes of transmission per orbit. Occultation exit will typically occur near the Martian north pole.

After the spacecraft exits the zone of Earth occultation and completes the radio science experiment, the DSN will require a short amount of time to gain a "signal lock" for modulated telemetry. This time will amount to a maximum of five minutes on the first observed orbit of a tracking pass, but drop to

Figure 6-6: *Typical Profile for Playback Orbit*



one minute on each subsequent orbit. Then, the spacecraft will be free to downlink its recorded science and engineering data to Earth as it orbits from north to south around the "front" side of Mars.

Another radio occultation experiment will occur on every playback orbit as the spacecraft approaches the Martian South Pole, but before it passes behind Mars relative to the Earth and enters the occultation zone. Like the occultation exit experiment previously described, the entry experiment will consume about five minutes of available time for transmission of recorded data.

All of this information leads to the fact that the total available playback time on each orbit of a DSN tracking pass can be determined by taking the orbit period, and subtracting the time the spacecraft spends in Earth occultation, the time allocated for the radio-science experiment, DSN lockup time, solid-state recorder (SSR) management activities, and navigational uncertainty. With these subtractions, only 67.5 minutes per playback orbit out of the total 117.64 will be available for data return on a worst-case orbit. Therefore, over a period of five playback orbits that elapse during a standard 10-hour DSN tracking pass, a theoretical total of 337.5 minutes (five 67.5-minute segments) will be available for downlink (worst-case design constraint).

However, several other design constraints will reduce the true amount of available downlink time per tracking pass from the 337.5-minute theoretical total. First, for the purpose of design simplicity and operational cost savings, the playback command script will be programmed to downlink recorded data only in discrete 67.5-minute segments. Second, DSN tracking stations will require five minutes to gain a "signal lock" on modulated telemetry the first time the station attempts to listen to telemetry on any given tracking pass. Therefore, in order to avoid potential loss of recorded data due to this initial five minute lock-up period, the playback of recorded data will not begin until the second 67.5-minute segment of the DSN tracking pass. Satisfying this constraint will result in only four guaranteed, whole, discrete 67.5-minute segments (270 minutes total) available for playback per 10-hour tracking pass.

The net result is that the playback rate must exceed the data record rate by a 5.33 to 1.00 ratio (24 hours of data in 270 minutes). MGS will utilize playback rates 21.3, 42.7, and 85.3 ksps that correspond to record rates of 4.0, 8.0, and 16.0 ksps, respectively. These rate pairs were selected to match the 5.33 to 1.00 ratio, and to cover the range of expected telecommunications capability as the Earth-to-Mars distance varies over the mapping phase. At maximum Earth-to-Mars range, the link will only support the 21.3 ksps playback rate. At other times, the operations team will select the highest downlink rate supportable by the link margin.

If the DSN tracking pass begins when the spacecraft is in view of the Earth, then the pass will consist of one partial segment at the beginning, four whole segments in the middle, and one partial segment at the end. However, if the tracking pass begins during Earth occultation, then the pass will consist of five whole, unbroken segments. Either way, the beginning of the first segment (either whole or partial depending on the exact timing) will serve as the “lock-up” period, and playback will not begin until the second segment.

In order not to “waste” the first segment of a tracking pass, the spacecraft will downlink realtime S&E1 data during that time. This data will be the same as that being recorded for the next day’s playback. The first five minutes of the realtime data will serve as a “throw-away filler” for the initial DSN signal “lock-up” period. Any additional realtime data returned during the first downlink segment of the pass will provide engineering telemetry for the ground monitoring teams to assess the current spacecraft “health” and status. This data is important because the engineering data downlinked during the playback session will represent the spacecraft status up to one day prior.

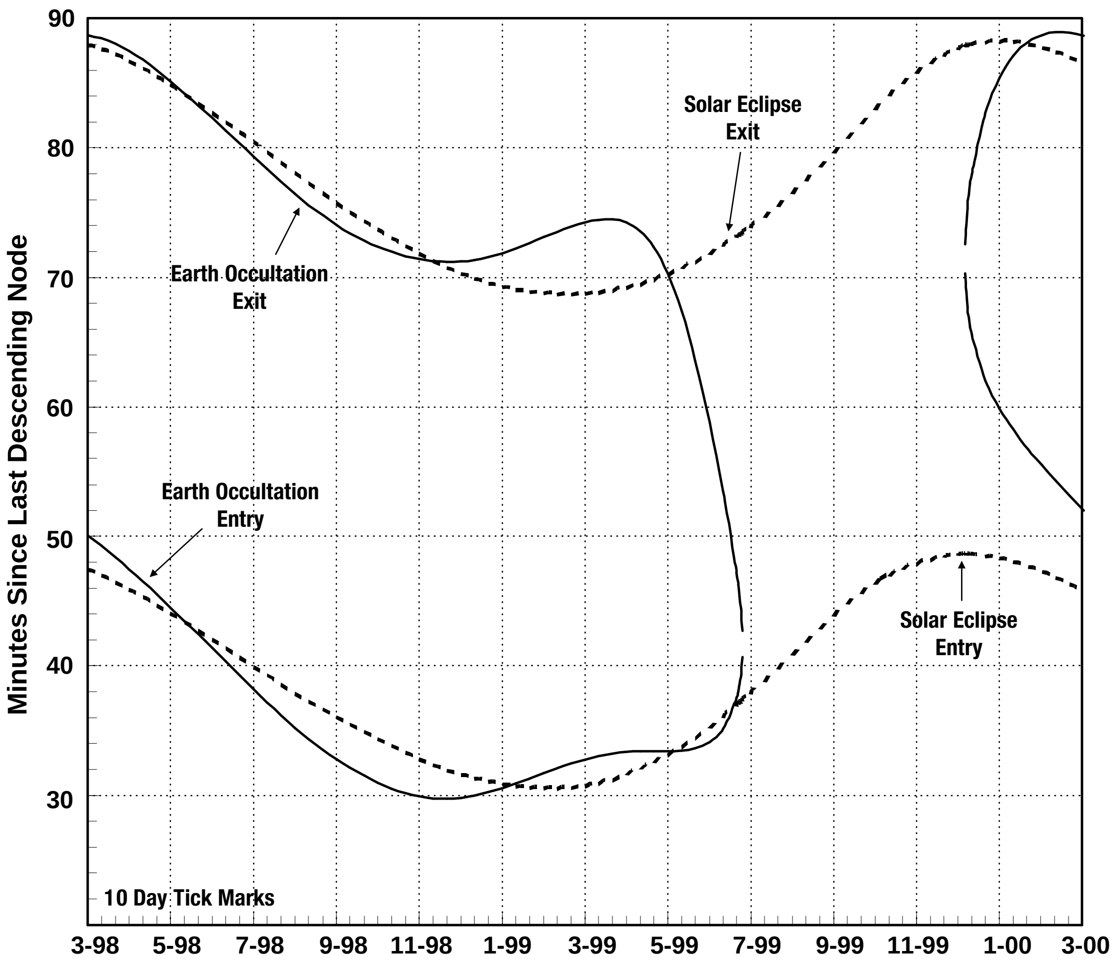
From 19 June 1999 to 25 November 1999, the MGS mapping orbit will appear “face on” relative to Earth, and no periods of Earth occultation will occur. However, such a geometry will not substantially increase the amount of transmit time available per orbit because of solar eclipse periods. During times when the spacecraft passes through Mars’ shadow, the batteries will provide power to run the electronics. Under the current design, running the transmitter throughout the solar eclipse period will drain the batteries to an unacceptably low depth of discharge level. Therefore, the data playback will still need to be broken up into the multiple 67.5-minute segment scheme previously discussed.

Autonomous Eclipse Management (AEM)

During a DSN tracking pass, the MGS spacecraft will use a capability called “autonomous eclipse management” (AEM) to trigger the execution of a command script that will perform the radio science and playback of recorded data. AEM takes the form of an algorithm that resides in the flight software power task. This algorithm will utilize power subsystem telemetry outputs (short circuit current, and solar array and battery currents) in a majority vote scheme to detect transitions into and out of eclipse to an accuracy within 12 seconds. The major advantage of using AEM to assist in autonomously returning recorded data will be manifested in significant savings in operational planning time and cost due to a reduction in the number of data playback command scripts needed over the mission’s mapping phase.

One of the key aspects of using AEM to trigger a playback script involves determining when to begin the playback because downlink cannot occur during periods in the orbit when Mars occults the Earth. When the spacecraft flies through the zone of Earth occultation behind Mars, it will also fly through the zone of solar eclipse (see Figure 6-6 and Figure 6-7). The reason that these zones overlap is that the vectors from Mars to the Sun and Earth point in approximately the same direction. However, the durations of Earth occultation and solar eclipse, as well as the relative entry and exit times will vary throughout the mission’s mapping phase. As shown in Figure 6-7, the occultation ingress and egress times may occur before or after the eclipse ingress and egress times, respectively. This fact is significant because choosing eclipse exit as the script trigger may result in a late playback start because occultation exit will have already occurred during many days of the mapping phase. Therefore, eclipse entry must serve as the triggering mechanism for the playback script.

Figure 6-7: *AEM Timing Relationship Between Earth Occultation and Solar Eclipse*

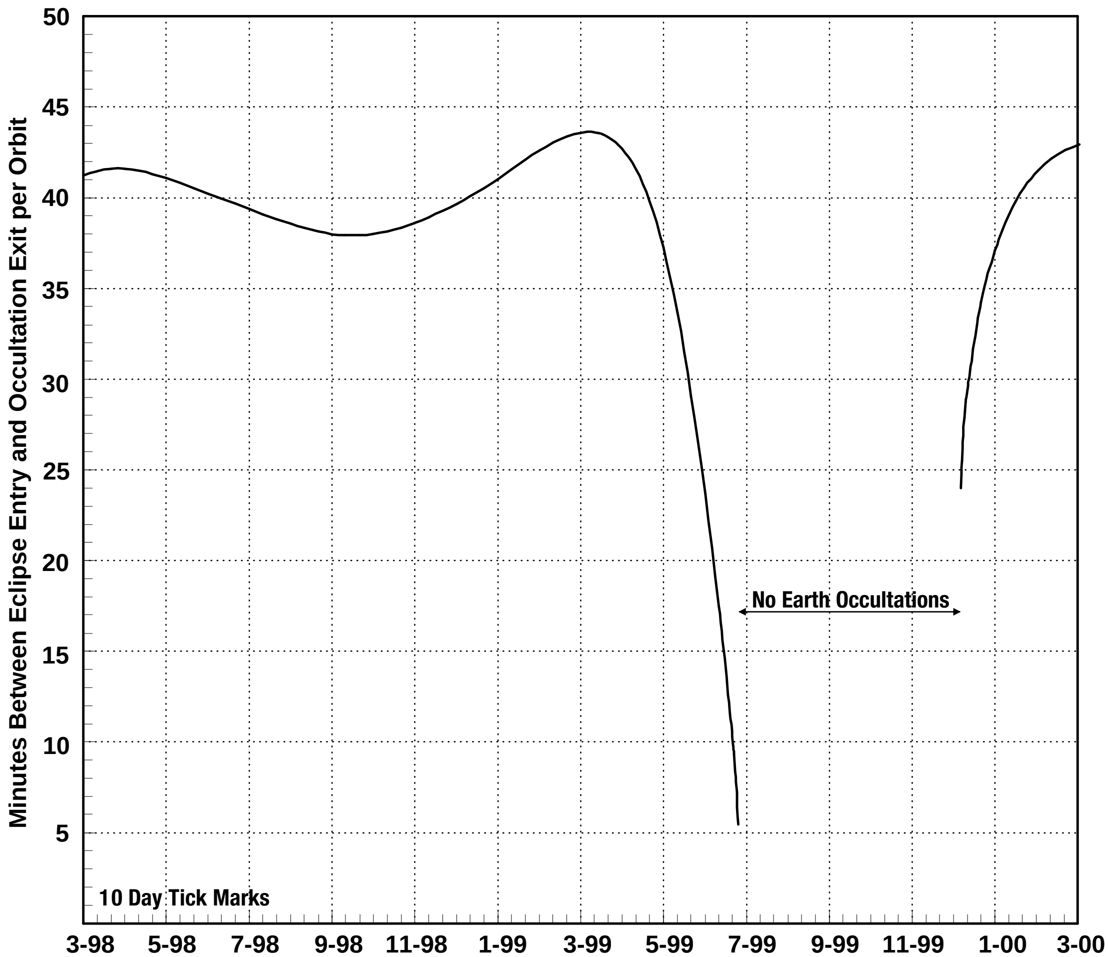


Due to the timing variation between eclipse entry and occultation exit, the first command that the spacecraft will execute from the solar eclipse ingress triggered script will be to wait until two minutes before Earth occultation exit. This “delay” time will be implemented as a variable parameter loaded from the ground. Figure 6-8 shows the amount of time that elapses between eclipse ingress and occultation egress as a function of time during the mapping phase. Under normal circumstances, the delay parameter will be updated about once every six weeks.

When the “delay” expires, the spacecraft will turn on its transmitter to perform the radio science atmospheric occultation experiment. “Transmitter on” will occur about two minutes before the spacecraft exits Earth occultation to allow enough time for the downlink signal to stabilize prior to the start of radio science. The two minutes allocated for transmitter warm-up will be subject to change after integration and test of the telecommunications subsystem and during flight as actual performance data becomes available.

After transmitter warm-up, the command script will configure the telecommunications subsystem for radio science by turning off telemetry modulation and switching to one-way, non-coherent mode using the Ultra-Stable Oscillator (USO) as the downlink frequency source. This configuration will maximize the signal strength and stability as it passes through the Martian atmosphere on the way to Earth. The radio science experiment will last from Earth occultation exit to when the spacecraft has moved to a point in the orbit where the radio signal no longer passes through Mars’ atmosphere on its way to Earth (defined at an altitude of 200 km above the Martian surface). This time will be determined from navigation predicts and will vary throughout the mapping phase.

Figure 6-8: ***Elapsed Time Between Eclipse Ingress and Occultation Egress***



Playback of recorded S&E1 data from the appropriate SSR will begin immediately after the end of radio science and will last for 67.5 minutes. During this time, downlink will be configured for telemetry modulation in two-way coherent mode. After 67.5 minutes, the command script will reconfigure the downlink for another radio science occultation experiment as the spacecraft begins to pass behind Mars relative to the Earth. Simultaneously, the spacecraft will wait for the detection of solar eclipse entry to trigger the playback command script for the next orbit of the DSN tracking pass.

6.3.2 Data Collection Strategy for Realtime Data (S&E2)

The MGS mission will also utilize data rates of 40 and 80 ksps to permit the realtime return of high-bandwidth data that would otherwise be constrained to lower record rates (4, 8, or 16 ksps). Approximately every third day during the mapping phase, an additional 10-hour tracking pass (as compared to the daily tracking passes) will allow realtime data return. This mode is called S&E2, and only the Mars Orbiter Camera (MOC) and Thermal Emissions Spectrometer (TES) will utilize this realtime mode.

During periods of S&E2 operations, the solid state recorders will continue to record data in the S&E1 mode for daily playback (described in the previous section) to maintain continuous observations for Magnetometer (MAG) and Mars Orbiter Laser Altimeter (MOLA) data. Another reason for simultaneous S&E1 recording is that realtime data transmission cannot occur in the Earth occultation zone, primarily on the night side of the planet. However, the bit allocations for the different instruments within the S&E1 data stream will be slightly different when transmitting S&E2 (on the “day side” of the planet) than during normal S&E1 record operations. When the spacecraft flies over the night side of Mars, the data collection mode will temporarily switch back to normal S&E1 for the duration of flight through the Earth occultation zone. Detailed information regarding the bit allocations for MOC and MOLA within the S&E2 data stream and the modified bit allocations within the S&E1 data stream appear in Appendix A.

Although a 10-hour view period will be requested from the DSN for the every third day realtime S&E2 track, an eight-hour period will suffice when either the link margin will not support a 10-hour DSN pass, or the physical viewing geometry (horizon to horizon) will not support a 10-hour pass. The reason is that the return of realtime data will not require four discrete 67.5 minute transmit segments as in the case of S&E1 playback. Using three discrete segments, plus two partial segments, will work as long as the total realtime transmit time totals 270 minutes.

Similar to the strategy used to control the return of S&E1 data, S&E2 return will also utilize the spacecraft's autonomous eclipse management (AEM) capability. The scripts for both types of data modes will resemble each other. The main difference will be that instead of commanding playback from the SSRs after radio science at occultation egress, the spacecraft will downlink realtime telemetry.

6.3.3 Science Campaign Schedule

Science campaigns will consist of 10 individual instances of intensive periods of continuous, real-time S&E2 data return to observe Mars during periods when unique surface and atmospheric conditions occur due to seasonal changes, during periods of special orbit geometry, and for spatial or global coverage issues (see Table 6-3). Nine of the 10 campaigns will last for 88 revolutions over a period of roughly seven sols to provide global coverage for a complete orbit repeat cycle. One of the campaigns will last for four repeat cycles over a time period of 29 days.

Table 6-3: ***Science Campaign Schedule***

Mission Dates	Code	L_s	Disciplines	Observation Targets and Comments
15-Mar-98 to 13-Apr-98	A	300^o	Atmosphere #1 Color Imaging #1 Global Topography #1	**Northern WInter / Southern Summer** CO_2 clouds in the northern polar regions, start of dust and winter storms, high-resolution topography map for MOLA.
29-Jun-98 to 6-Jul-98	B	354^o	Atmosphere #2 Color Imaging #2	**Northern Winter / Southern Summer** Residual southern polar cap, close of dust storm period.
26-Oct-98 to 2-Nov-98	C	47^o	Atmosphere #3 Color Imaging #3 Diametric Occultation #1 Gravity #1	**Northern Spring / Southern Fall** Mid-spring water sources and CO_2 clouds in the southern polar regions. Earth is in the plane of the mapping orbit for gravity campaign and diametric occultation.
5-Jan-99 to 12-Jan-99	D1	80^o	Geodesy Look 1, Pass 1	**Start of Southern Winter** Atmosphere in both hemispheres is colorist at this time for geodesy campaigns. Earth is in the plane of the mapping orbit for the diametric occultation. Each geodesy "week" must be separated by multiples of 88 orbits.
20-Jan-99 to 27-Jan-99	D2	87^o	Geodesy Look 2, Pass 1	
3-Feb-99 to 10-Feb-99	D3	94^o	Geodesy Look 1, Pass 2	
18-Feb-99 to 25-Feb-99	D4	101^o	Geodesy Look 2, Pass 2 Diametric Occultation #2	
3-May-99 to 10-May-99	E	137^o	Atmosphere #4 Color Imaging #4 Gravity #2	**Northern Summer / Southern Winter** Atmosphere profiles near the equator and water transport across the equatorial zone.
4-Oct-99 to 11-Oct-99	F	213^o	Color Imaging #5 Gravity #3	**Start of Northern Fall** Earth out of mapping orbit plane angle is at a maximum.
13-Dec-99 to 20-Dec-99	G	263^o	Atmosphere #5	**Start of Southern Summer** CO_2 clouds in northern polar regions and start of dust and winter storms.

As shown in Table 6-3, the 29-day campaign and each of the week-long campaigns will be dedicated to one or more disciplines in the fields of atmospheric studies, diametric occultations, global color imaging, geodesy, and Martian gravity field measurements in the form of radiometric data from radio science. In general, the science campaign schedule calls for as many of the “sub-campaigns” corresponding to the various individual disciplines to occur at the same time in order to minimize the amount of continuous tracking requested from the Deep Space Network (DSN).

The 29-day science campaign will start at the beginning of the mapping on 15 March 1998, and will primarily accommodate the production of the high-resolution MOLA topographic map of Mars. This production will provide an early, high-quality product from the MOLA investigation team and will ensure useful scientific results in case of laser degradation on the instrument. Although MOLA does not use real-time S&E2 downlink, continuous DSN tracking for this campaign is necessary in order to provide post-navigation reconstruction of the orbit at a fidelity high enough to achieve the necessary vertical accuracy for the construction of the map. Without continuous tracking (10 hours vs. 24 hours per day), the errors introduced into the modeling process due to atmospheric drag and solar radiation may be increased by a factor of five. In addition, the month-long campaign will enable the return of a high volume of data from all of the instruments at the beginning of mapping. Such a scenario is highly desirable because at the start of mapping, the spacecraft will have already spent nearly seven months at Mars without significant return of science data due to orbit insertion, aerobraking, and transition to mapping.

During each day of a campaign, the current plan involves transmitting S&E2 realtime for two-thirds of the day, and then downlinking the previous 24 hours of recorded S&E1 data in the remaining one-third of the day. In a manner similar to the strategy used for collecting realtime data during the “extra” pass every third day as described previously, the solid state recorders (SSRs) will continue to record data in the standard S&E1 mode for daily playback while the spacecraft simultaneously transmits the realtime, S&E2 data stream back to Earth. Because this daily playback will require eight hours to complete, realtime data return during science campaigns will be limited to eight out of the 12 Martian orbits that the spacecraft completes in a typical day.

6.3.4 Data Rates and Mapping Sequences

Daily playback of data recorded on the solid state recorders will utilize downlink transmission rates of 21.3, 42.7, and 85.3 ksps (corresponding to record rates of 4.0, 8.0, and 16.0 ksps, respectively). These rate pairs were selected to match the 5.333 to 1.0 playback-to-record ratio constraint, and to cover the range of expected telecommunications capability as the Earth-to-Mars distance varies over the mapping phase. Specifically, the spacecraft will always utilize the fastest playback rate supportable for an entire DSN station pass.

The choice of whether to utilize either the 40-ksps or 80-ksps transmission rate for realtime data return will also depend on choosing the fastest rate supportable for an entire DSN station pass. For example, on days when the link margin can only support the 42.7-ksps playback rate, the 40 ksps rate will be used for realtime operations. The 80-ksps rate will be reserved for realtime data return on days when the link margin and DSN view geometry of Mars can support the 85.3-ksps playback rate.

Between the dates of 15 March 1998 and 25 October 1998, the link margin will only support the 21.3-ksps data rate because Mars will be at maximum range relative to the Earth. Although this performance limitation will not impact the playback of recorded data, it will preclude using the 40-ksps rate for the return of realtime data under normal, two-way coherent communication conditions. During this time period, the DSN will drop the uplink carrier during realtime data return periods. This scheme will boost the signal gain to a level that can support the 40-ksps rate by removing the antenna transmitter noise from the link equation. Unfortunately, lack of an uplink will eliminate the possibility of spacecraft and instrument commanding during realtime data return periods. For these seven months, commanding will only be possible during the daily 10-hour DSN pass used for the playback of recorded data during normal mapping operations, or during the one-third of the day allocated to playback during science campaigns.

Table 6-4 shows the dates of usage for the different data rates over the course of the entire mapping phase. As shown in the table, the mapping phase of the mission will consist of 19 distinct time segments named M1 through M19. Each segment is called a “mapping sequence” and will last anywhere from 28 to 42 days. Prior to the start of the sequence, all of the basic spacecraft commands for that time period will be uplinked in a single batch. Consequently, the transmission rates used for playback and realtime data will not change during the course of a single sequence.

The boundary spacing of 28 to 42 days between sequences will minimize the number of command loads generated during mapping without sacrificing the prediction accuracy of the times that spacecraft events must occur relative to orbit events. However, commanding of science instruments may take place at any time during a sequence.

Table 6-4: ***Data Rate Usage During Mapping***

Dates	ID Code	Range	Playback Mode (PB)	Realtime Mode (RT)	Elevation Mask
15-Mar-98 to 13-Apr-98	M1	2.43 AU	21.3 ksps	40 ksps	25^o / 35^o (8^o for PB)
14-Apr-98 to 03-May-98	M2	2.47 AU	21.3 ksps	40 ksps (one way)	12^o / 16^o
04-May-98 to 21-May-98	M3	2.50 AU	n/a (solar conjunction)	n/a (solar conjunction)	12^o / 16^o
22-May-98 to 28-Jun-98	M4	2.52 AU	21.3 ksps	40 ksps (one way)	13^o / 17^o
29-Jun-98 to 09-Aug-98	M5	2.52 AU	21.3 ksps	40 ksps (one way)	13^o / 17^o
10-Aug-98 to 20-Sep-98	M6	2.46 AU	21.3 ksps	40 ksps (one way)	11^o / 16^o
21-Sep-98 to 25-Oct-98	M7	2.30 AU	42.7 ksps	40 ksps	21^o / 29^o
26-Oct-98 to 29-Nov-98	M8	2.08 AU	42.7 ksps	40 ksps	12^o / 17^o
30-Nov-98 to 03-Jan-99	M9	1.80 AU	42.7 ksps	40 ksps	07^o / 09^o
04-Jan-99 to 31-Jan-99	M10	1.46 AU	85.3 ksps	80 ksps	12^o / 16^o
01-Feb-99 to 14-Mar-99	M11	1.18 AU	85.3 ksps	80 ksps	06^o / 07^o
15-Mar-99 to 25-Apr-99	M12	0.79 AU	85.3 ksps	80 ksps	06^o / 06^o
26-Apr-99 to 06-Jun-99	M13	0.68 AU	85.3 ksps	80 ksps	06^o / 06^o
07-Jun-99 to 18-Jul-99	M14	0.92 AU	85.3 ksps	80 ksps	06^o / 06^o
19-Jul-99 to 29-Aug-99	M15	1.17 AU	85.3 ksps	80 ksps	06^o / 07^o
30-Aug-99 to 11-Oct-99	M16	1.41 AU	85.3 ksps	40 ksps	10^o / 14^o (6^o for RT)
12-Oct-99 to 21-Nov-99	M17	1.63 AU	42.7 ksps	40 ksps	06^o / 07^o
22-Nov-99 to 02-Jan-00	M18	1.85 AU	42.7 ksps	40 ksps	07^o / 10^o
03-Jan-00 to 31-Jan-00	M19	2.01 AU	42.7 ksps	40 ksps	10^o / 14^o

In Table 6-4, the column that contains elevation mask information contains two numbers that specify the minimum elevation angle of Mars, as seen from a 34m HEF antenna, that can support the data rates listed for the given sequence. The first number specifies the minimum elevation at Goldstone, while the second number specifies the same information for either Madrid or Canberra. In all cases, the minimum elevation angle required at Goldstone to support a given data rate is less than or equal to that at the other two DSN sites. The reason is that air around Goldstone contains less water vapor on average than either Madrid or Canberra. Therefore, received signal experiences less attenuation and produces a stronger link margin at Goldstone.

The elevation angle numbers presented in Table 6-4 for each mapping sequence represent values averaged across the entire sequence. Consequently, the actual values at any given instant in time will be slightly different. For the purposes of DSN scheduling and conflict resolution with other projects, the elevation masks will be computed on a week by week basis and supplied to the DSN at the appropriate times.

6.4 OTM Sequencing

Orbit trim maneuvers (OTMs) during the mapping phase will be performed for the purpose of controlling the spacing of the repeating ground tracks, and for correcting the orbit eccentricity and inclination. It is currently anticipated that OTMs will be scheduled approximately once every month. These maneuvers will normally be placed near periapsis, although they will occasionally occur near apoapsis. However, OTMs designed specifically for inclination control must be executed near the equator.

Four-hour windows will be provided in the mapping sequences for the OTMs. These windows will allow the maneuver time to be selected from one of two different orbits in the event that large gravity anomalies are present on one of the orbits. Spacecraft and payload activities which conflict with the maneuver will be kept out of the window.

6.5 Solar Conjunction

The mapping phase will be interrupted for a period centered about 12 May 1998 at 20:00 UTC. At that time, Mars will pass through solar conjunction as seen from the Earth (minimum Sun-Earth-Mars angle of 0.04 degrees). Due to solar interference, both the uplink and downlink radio signals will be degraded for a period of several days on each side of the conjunction. Based on current analysis, it is assumed that the uplink signal will be unacceptably degraded for commanding when SEM angle is less than 2^{o}.

This uplink degradation constraint will result in a 17-day command moratorium starting on 4 May 1998 and ending on 21 May. The spacecraft will operate autonomously during this time period and will maintaining all of its normal engineering functions without commanding over this 17-day period. Throughout most of solar conjunction, it will be highly unlikely that telemetry will be successfully received at Earth even when using the slowest S&E1 playback rate of 21.3 ksps. Consequently, solid-state recorder playbacks will not occur during this time period.

The current plan is to use a special AEM script that will provide realtime spacecraft telemetry and navigation Doppler data during a single DSN tracking pass each day. This strategy will allow for daily health assessment of the spacecraft and the payload, provide for radiometric monitoring of the orbit, and possibly allow for the return of limited science observations. Unfortunately, realtime performance at the lowest science collection rate of 4.0 ksps may not be achievable throughout all of the solar conjunction period. Consequently, a hybrid strategy may be used. For the first ten minutes of each orbit, engineering-only telemetry will be returned at 250 bps. For the rest of the orbit, 4.0 ksps S&E-1 will be returned in real-time. This strategy will permit the spacecraft to attempt to return science data and enhance the chance that some spacecraft engineering data will reach Earth.

Radio science atmospheric-occultation measurements will also be performed during solar conjunction. During the radio-science segments of each orbit, the transponder will be set in 1-way mode with telemetry and ranging modulation off. During the rest of the orbit, the transponder will be configured with telemetry and ranging on, and with 2-way coherent tracking enabled. As the conjunction geometry becomes more extreme, the uplink carrier may not be transmitted from the ground because removing the ground transmitter from the link and utilizing the “listen-only” mode will boost the effective gain of the downlink signal.

The data return profile described above will be repeated for each orbit during solar conjunction, despite the fact that tracking requirements call for only one 34m HEF pass per day. If anomalies or any other circumstances indicate that additional coverage is warranted, downlink from the spacecraft will be available without emergency commanding. When commanding becomes possible after conjunction, normal mapping data return will begin.

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Appendix A - Data Rate Modes

This appendix describes the modes that the MGS spacecraft will use for the collection of science and engineering data. In the two tables in this appendix, “bps” stands for bits per second and “sps” stands for symbols per second. A symbol is essentially a Reed-Solomon encoded (250:218 ratio) data bit. Therefore, it takes approximately 1.147 bits of storage space to encode one bit of raw data.

The data rates corresponding to the science instruments represents an average rate because the Payload Data Subsystem (PDS) does not receive data continuously. Each one of the science instruments generates data in discrete packets collected by the PDS at regular intervals determined by the internal software. The packets from each instrument will include identification and timing information in addition to the raw science data. In addition to collecting instrument packets, the PDS will also gather spacecraft engineering packets and engineering packets from itself. After collecting the packets, the PDS will format them into transfer frames 10,000 symbols (8,720 bits) in length, including a header 80 bits in length.

Table A-1 lists the rates for the science instruments and engineering data streams corresponding to S&E1 operations. In this mode, the data can either be stored on the solid state recorders (SSRs) for later playback, or downlinked back to Earth in realtime. In S&E2 operations as shown in Table A-2, the PDS will simultaneously generate two data streams. The first data stream (S&E2) will be downlinked back to Earth in realtime. The second data stream (modified S&E1) will be stored onto the solid state recorders (SSRs) for later playback. In Table A-2, the numbers in parenthesis correspond to the modified S&E1 data rate. All numbers in Tables A-1 and A-2 are in bits per second (bps) unless otherwise specified.

Both tables in this appendix show data rates for GRS and PMIRR. These two scientific instruments were flown on the Mars Observer mission, but were not included as part of the MGS payload due to mass considerations. When MGS inherited its PDS from Mars Observer, the data rates for these two instruments were not reallocated.

Table A-1: ***S&E1 Data Modes***

	LRC	MRC	HRC	MR	EOM
Description	Low Record	Medium Record	High Record	Mars Relay	Eng. Only
Total Symbol Rate (sps)	4000	8000	16000	4000	4000
Total Data Bit Rate	3488	6976	13952	3488	3488
MAG/ER Data Rate	324	648	1296	324	0
MOLA Data Rate	618	618	618	618	0
MOC Data Rate	700	2856	9120	2670	0
TES Data Rate	688	1664	1664	0	0
Unallocated (GRS)	665	665	665	0	0
Unallocated (PMIRR)	156	156	156	0	0
S/C Engineering	256	256	256	256	256
PDS Engineering	48	48	48	48	3072
T/F Header	32	64	128	32	32
Filler	1	1	1	2	128

Table A-2: ***Combined S&E2 and S&E1 Data Modes***

	RTL		RTM	
Description	Real Med.	Low Rec.	Real Med.	Med. Rec.
Total Symbol Rate (sps)	40000	(4000)	40000	(8000)
Total Data Bit Rate	34880	(3488)	34880	(6976)
MAG/ER Data Rate	0	(324)	0	(648)
MOLA Data Rate	0	(618)	0	(618)
MOC Data Rate	29260	(1388)	29260	(4520)
TES Data Rate	4992	(0)	4992	(0)
Unallocated (GRS)	0	(665)	0	(665)
Unallocated (PMIRR)	0	(156)	0	(156)
S/C Engineering	256	(256)	256	(256)
PDS Engineering	48	(48)	48	(48)
T/F Header	320	(32)	320	(64)
Filler	4	(1)	4	(1)

Table A-2: ***Combined S&E2 and S&E1 Data Modes (continued)***

	RTH1		TST	
Description	Real Med.	High Rec.	Real High	High Rec.
Total Symbol Rate (sps)	40000	(16000)	80000	(16000)
Total Data Bit Rate	34880	(13952)	69760	(13952)
MAG/ER Data Rate	0	(1296)	0	(1296)
MOLA Data Rate	0	(618)	0	(618)
MOC Data Rate	29260	(10782)	63808	(10782)
TES Data Rate	4992	(0)	4992	(0)
Unallocated (GRS)	0	(665)	0	(665)
Unallocated (PMIRR)	0	(156)	0	(156)
S/C Engineering	256	(256)	256	(256)
PDS Engineering	48	(48)	48	(48)
T/F Header	320	(128)	640	(32)
Filler	4	(3)	16	(3)

Appendix B - Payload Data Sheet

Table B-1: ***Payload Data Sheet (1 of 2)***

	MAG/ER	MOC	MOLA
Instrument Name	Magnetometer and Electron Reflectometer	Mars Orbiter Camera	Mars Orbiter Laser Altimeter
Principal Investigator Instrument Manager	M. H. Acuna J. Scheifele	M. C. Malin G. E. Danielson	D. E. Smith B. L. Johnson
Build Source	Mars Observer Spare	Mars Observer Spare	Assemblies
Mass	5.3 kg	21 kg	25.9 kg
Power (Peak / Average)	4.63 / 4.63 W	29.75 / 10.50 W	34.94 / 30.94 W
Operating Voltage	28 ± 0.56 V	28 ± 6 V	28 ± 2 V
Operating Temp. Limit	-10^o C to 50^o C	-28^o C to -2^o C (NA) -62^o C to -24^o C (WA)	-20^o C to 30^o C
Non-Operating Temp. Limit	-40^o C to 75^o C	-30^o C to 40^o C	-30^o C to 40^o C
Field of View	4π steradian (MAG) 360^o x 14^o (ER)	140.2^o x 3^o (WA) 0.44^o (NA)	0.85 mrad
Resolution	200 km	250 meters/pixel (WA) 1.4 meters/pixel (NA)	160 meters horizontal 2 meters vertical
Approximate Size	20 x 17.6 x 15 cm	86 x 45 x 40 cm	71 x 58 x 62 cm

Table B-2: ***Payload Data Sheet (2 of 2)***

	MR	TES	USO
Instrument Name	Mars Relay	Thermal Emission Spectrometer	Ultra-Stable Oscillator
Principal Investigator Instrument Manager	J. Blamont A. Ribes	P.R. Christensen G. Mehall	G.L. Tyler (team leader) C.L. Hamilton
Build Source	Print	Assemblies	Mars Observer Spare
Mass	8.0 kg	14.1 kg	1.3 kg
Power (Peak / Average)	13 / 13 W (includes MOC)	18.20 / 15.55 W	4.5 (warm-up) / 3.0 W
Operating Voltage	10 ± 0.50 V (needs MOC)	26 to 32 V	24 to 30 V
Operating Temp. Limit	-10^o C to 40^o C	-20^o C to 30^o C	-20^o C to 30^o C
Non-Operating Temp. Limit	-30^o C to 40^o C	-20^o C to 50^o C	-30^o C to 40^o C
Field of View	130^o	0.95^o x 1.43^o	n/a
Resolution	Limb to Limb	3 km per detector	n/a
Approximate Size	24 x 25 x 19 cm	36 x 24 x 40 cm	16 x 9 x 10 cm

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Appendix C - Compliance With Project Requirements

This appendix details the compliance of the Mars Global Surveyor mission design with the Mission Requirements Document (MRD) and the Instrument Description and Science Requirements Document (IDSRD).

C.1 Compliance With the Mission Requirements Document

Table C-1 gives an assessment of compliance with the MRD, based on the September 1995 version of that document. As shown in the table, all requirements are met with the current mission plan.

C.2 Compliance With the Science Requirements Document

Table C-2 gives an assessment of compliance with the IDSRD, based on the February 1995 version of that document and shows that all requirements are met with the current mission plan. Many of the science desires, such as special observations during the cruise and orbit insertion phases, are being evaluated by the project. References are given in the table to specific sections in the Mission Plan that discuss each requirement.

Table C-1: ***MRD Requirements Compliance***

MRD #	MRD Requirement	Achieved?	Reference and Comments
3.1.1	Launch Opportunity	Yes	Section 3.3 - Launch Strategy
3.1.2	Launch Vehicle	Yes	Section 3.1 - Launch Vehicle Description
3.1.3	Launch Period	Yes	Section 3.3 - Launch Strategy
3.1.4	NASA Planetary Protection	Pending	Planetary Protection Plan (542-402)
3.1.5	Cruise and Orbit Insertion Science	Yes	Section 4.3 - Inner Cruise Activities Section 4.4 - Outer Cruise Activities Section 5.3 - Transfer to Mapping
3.1.5.1	Payload Engineering Checkout	Yes	Section 4.3 - Inner Cruise Activities
3.1.5.2	MOC Post Bakeout Focus Check	Yes	Section 4.4 - Outer Cruise Activities
3.1.5.3	MAG Calibration	Yes	Section 4.4 - Outer Cruise Activities
3.1.5.4	MOC Pre-MOI Mars Approach Observations	Yes	Section 4.4 - Outer Cruise Activities
3.1.5.5	Science Observations During the Orbit Insertion Phase	Yes	Section 5.1 - Mars Capture Subphase Section 5.2 - Aerobrake Subphase MAG data may not be achievable during walkout
3.1.5.6	MOC Star Images Before Mapping Deployment	Yes	Section 5.3 - Transfer to Mapping Subphase
3.1.6	Mapping Orbit Design	Yes	Section 6.1 - Mapping Orbit Design Orbit has 7 sol repeat, index alt. of 378 km
3.1.7	Gravity Calibration Period	Yes	Section 5.3 - Transfer to Mapping Subphase
3.1.8	S/C and Instrument Checkout Period	Yes	Section 5.3 - Transfer to Mapping Subphase Checkout period of 10 days
3.1.9	Mapping Phase Commencement	Yes	Section 5.3 - Transfer to Mapping Subphase Section 6.0 - Mapping Phase
3.1.10	Mapping Phase	Yes	Section 6.0 - Mapping Phase
3.1.11	Relay Phase	Pending	Information to be updated as Mars-98 mission design reaches maturity
3.1.12	Science Data Return	Yes	Section 6.3 - Data Collection Strategy
3.1.13	Radio Science Data	Yes	Section 6.3 - Data Collection Strategy
3.1.14	Solar Conjunction Command Moratorium	Yes	Section 6.5 - Solar Conjunction 17-day command moratorium at conjunction
3.1.15	DSN Usage	Yes	Section 2.6 - DSN Utilization Consistent with DMR and MRR documents
3.1.16 - 1	First Maneuver Constraint	Yes	Section 4.2 - Trajectory Correction Maneuvers
3.1.16 - 2	Maneuver Interval Constraint	Yes	Section 4.0 - Cruise Timeline Section 4.2 - Trajectory Correction Maneuvers Section 5.0 - Orbit Insertion Timeline Section 5.1 - MOI Maneuver Section 5.2 - Aerobrake Subphase
3.1.16 -3	Sun-safe Maneuver Direction Constraint	Yes	Section 4.2 - TCM Schedule/Implementation Section 5.2 - Aerobrake Subphase
3.1.17	Aerobraking	Yes	Section 5.2 - Aerobrake Subphase
3.1.18	S/C Overheating During Capture	Yes	Section 5.2 - Aerobrake Subphase

Table C-2: *IDSRD Requirements Compliance*

IDSRD #	IDSRD Requirement	Achieved?	Reference and Comments
3.2.2.1 (MAG1&2))	1-Day drift orbit with calibration data, 28-day minimum	Yes	Section 5.0 - Orbit Insertion Phase MGS does not have a 1-day drift orbit due to utilization of aerobraking to achieve mapping orbit. MAG data will be returned throughout most of the orbit insertion phase.
3.2.2.2 (MAG3)	Mapping orbit 350 ± 50 km	Yes	Section 6.1 - Mapping Orbit Design Orbit index altitude is 378 km
3.2.2.3 (MAG4)	Successive ground track separation 1/3 to 1/20 of orbit altitude	Yes	Section 6.1 - Mapping Orbit Design Orbit walk is about 1/6 (58.6 km)
3.2.2.3 (MAG5)	Repeat period not commensurable with solar rotation period of 26.354 Earth days	No	Requirement cannot be met with current project cost and staffing assumptions
3.2.2.4 (MAG6)	Minimum 10 mapping cycles	No	Requirement cannot be met with current project cost and staffing assumptions
3.2.2.5 (MAG7)	Mapping orbit fixed in local time	Yes	Section 6.1 - Mapping Orbit Design Orbit is Sun-synchronous at 2:00 p.m. mean
3.2.3.1 (MOC1)	Altitude above true surface must be less than 540 km	Yes	Section 6.1 - Mapping Orbit Design Apoapsis areographic altitude is 426.6 km
3.2.5.1 (MOLA1&2)	Mapping orbit inclination within 3^o of 90^o, difference between orbit periapsis and apoapsis less than 105 km, mean height less than 380 km	Yes	Section 6.1 - Mapping Orbit Design See description of orbital elements
3.2.6.1 (RS1)	Mapping orbit periapsis between 250 and 400 km	Yes	Section 6.1 - Mapping Orbit Design Periapsis altitude bounds are 360 to 389 km
3.2.6.2 (RS2&3)	Gravity calibration orbit with 7-10 day repeat cycle and continuous DSN tracking over one full cycle	Pending	Section 5.3 - Transfer to Mapping Subphase Gravity calibration scheduled between ABX and TMO, but with not with 7-10 day repeat cycle
3.2.7.1 (TES1)	Maintain altitude between 250 km and 2000 km	Yes	Section 6.1 - Mapping Orbit Design Orbit e = 0.007, index altitude = 378 km
3.2.7.2 (TES2)	Mapping cycles offset more than 9 km in consistent direction	No	Requirement cannot be met with current project cost and staffing assumptions
3.2.8.1 (IDS1&2)	Near-polar, Sun-synchronous, low-altitude orbit with short mapping cycle	Yes	Section 6.1 - Mapping Orbit Design Repeat cycle of 7 sols
3.2.8.4 (IDS3)	Near-circular, 2:00 p.m. local solar time equator crossing Sun-synchronous orbit, 350 km < h < 450 km	Yes	Section 6.1 - Mapping Orbit Design Mapping orbit is Sun-synchronous with respect to the ficti-tious-mean Sun
3.4.2 (MAG9)	Continuous data acquisition for 4 weeks in drift orbit prior to mapping orbit insertion	Yes	Section 5.0 - Orbit Insertion Phase MGS does not have a 1-day drift orbit due to utilization of aerobraking to achieve mapping orbit. MAG data will be returned throughout most of the orbit insertion phase.
3.4.2 (MAG10)	ER data collection during cruise and drift orbit	No for Cruise Yes for Orb. Ins.	Section 5.0 - Orbit Insertion Phase MGS does not have a 1-day drift orbit due to utilization of aerobraking to achieve mapping orbit. ER data will be returned throughout most of the orbit insertion phase.
5.1.2 (MAG24)	ER on if MAG calibration rolls are performed	Yes	Section 4.4 - Outer Cruise Activities MAG rolls will be performed if needed
5.1.2.2 (MAG25)	Continuous operation of MAG during mapping phase	Yes	Section 6.3 - Data Return Strategy Continuous data collection except solar conjunction
5.1.6.1 (RS12)	RS data required during cruise phase to test equip-ment, procedures and software	Yes	Section 4.3 - Inner Cruise Activities
5.1.6.3 (RS13)	S/C will transmit X-band signals whenever it is visible from Earth using project allocated DSN passes sub-ject to power constraints	Yes	Section 6.3 - Data Collection Strategy Radio Science planned during all mapping phase DSN passes

Table C-2: ***IDSRD Requirements Compliance***

IDSRD #	IDSRD Requirement	Achieved?	Reference and Comments
5.1.6.4 (RS18)	In-flight RS tests prior to data acquisition to determine warm-up time required	Yes	Section 4.3 - Inner Cruise Activities Section 4.4 - Outer Cruise Activities RS ORTs will be scheduled by the project and DSN to meet this requirement
5.1.8.4.2 (IDS4)	Acquire atmospheric and polar data in a nearly continuous manner at all times during a full Martian year	Yes	Section 6.1 - Mapping Orbit Design Section 6.3 - Data Collection Strategy Continuous data collection from all science instruments except during solar conjunction
5.3.6 (RS20)	RS observations include 20 seconds of data while spacecraft is geometrically behind Mars, data from surface to 200-km altitude, 100 seconds of data when transmission path is above 200 km from the surface, and appropriate timing pads for insurance.	Yes	Section 6.3 - Data Collection Strategy Occultation ingress and egress experiments planned for all DSN passes during mapping except periods of no Earth occultations
5.6.1.1 (RS24)	Regular, continuous, and complete tracking of the mapping orbit, using at least one 34m HEF DSN pass per day	Yes	Section 2.6 - DSN Utilization Minimum requirement satisfied, but coverage may not be regular and continuous

Appendix D - Mass and ΔV Budget Details

The tables and figures in this appendix show the technical details of the mission ΔV budget corresponding to the open and the close of the launch period (6 November 1996 and 25 November 1996, respectively). In addition, this appendix lists the spacecraft mass and propellant breakdowns corresponding to the ΔV budget.

The spacecraft's propellant load was developed from a single ΔV budget that incorporated elements both independent and dependent of the actual launch date. Some of the independent elements include mapping orbit maintenance, relay orbit maintenance, and aerobraking walk-in maneuvers. Examples of launch date dependent elements include the MOI burn, TCMs 1 and 2, and aerobraking main-phase and walk-out maneuvers.

Because the same propellant load will be used for all launch dates, the pre-launch ΔV reserves line item will be different for each launch date. The reason for this difference is that the propellant load will remain constant, but the ΔV requirements for the launch date dependent elements will change. Specifically, the bi-propellant ΔV requirements become more demanding toward the end of the launch period.

Table D-1: ***Spacecraft Mass Breakdown***

Spacecraft Component	Mass (kg)	
Total Spacecraft Injected Mass	**1062.1**	
Spacecraft Dry Mass (includes pressurant)	**674.7**	
Science Payload		75.90
Spacecraft		597.40
Pressurant (helium)		1.40
Spacecraft Propellant Load	**388.4**	
Usable N_2O_4		148.21 (86.03% tank fill level)
Usable N_2H_4		230.59 (95.50% tank fill level)
Unusable		7.58 (2% of total propellant fill)
Oxidizer Ballast		1.00
Load Error		0.99

Figure D-1: ΔV *Budget Used for Propellant Loading*

MARS GLOBAL SURVEYOR SPACECRAFT ΔV/PROPULSIVE PERFORMANCE
Mission ΔV Budget - Propellant Loading

MISSION ΔV BUDGET - ORR BASELINE (8/96)
SPACECRAFT INJECTED MASS (kg) 1062.10

		MONOPROP TRANSLATIONAL		BIPROP TRANSLATIONAL		MONOPROP ROTATIONAL		PROPELLANT BREAKDOWN			POSTBURN	EST. BURN DURATION		
MISSION PHASE	MANEUVER	Isp (s)	ΔV Reqd (m/s)	Isp (s)	ΔV Reqd (m/s)	Isp (s)	ΔV Reqd (m/s)	M-HYDRZ (kg)	B-HYDRZ (kg)	NTO (kg)	S/C MASS (kg)	MONOPROP #T	MONOPROP (s)	BIPROP (s)
CRUISE ΔV95	TCM-1/2	0.0	0.0	317.0	36.0	220.0	0.3	0.15	6.61	5.62	1049.72	-	-	57.7
	TCM-3/4	220.0	2.0	0.0	0.0	0.0	0.0	0.97	0.00	0.00	1048.75	4	117.9	-
	pre-Lnch ΔV Reserves	220.0	0.0	315.0	13.2	0.0	0.0	0.00	2.42	2.05	1044.28	-	-	21.0
CAPTURE	MOI (Rp = 3700 km, Per = 48 Hr)	0.0	0.0	317.0	975.1	220.0	5.9	2.85	151.98	129.18	760.27	-	-	1339.0
Aerobraking Walk-in	AB-1	0.0	0.0	315.0	7.5	220.0	0.1	0.04	1.00	0.85	758.39	-	-	8.6
	AB-i (i=2,4)	220.0	2.5	0.0	0.0	0.0	0.0	0.88	0.00	0.00	757.51	4	106.5	-
Aerobraking Main Phase	ABM Translation	220.0	5.0	0.0	0.0	0.0	0.0	1.75	0.00	0.00	755.76	-	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	5.0	2.03	0.00	0.00	753.73	-	-	-
Aerobraking Walk-out	ABM Translation	220.0	20.0	0.0	0.0	0.0	0.0	6.95	0.00	0.00	746.78	-	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	30.0	11.93	0.00	0.00	734.85	-	-	-
Aerobraking Contingency	Aerobraking Pop-Up	220.0	2.5	315.0	14.5	0.0	0.0	0.85	1.86	1.58	730.56	-	-	15.9
	post-Lnch ΔV Reserves	220.0	0.0	315.0	0.0	0.0	0.0	0.00	0.00	0.00	730.56	-	-	-
Transition To Mapping	ABX	0.0	0.0	317.0	60.2	220.0	0.4	0.14	7.57	6.44	716.41	-	-	66.3
	TMO	0.0	0.0	317.0	16.1	220.0	0.1	0.03	2.00	1.70	712.68	-	-	17.5
	OTM-1 (Frzn)	0.0	0.0	315.0	7.5	0.0	0.0	0.00	0.93	0.79	710.95	-	-	8.2
MAPPING Drag Dens 95%	OTM (Drag/GTE)	220.0	3.9	0.0	0.0	0.0	0.0	1.28	0.00	0.00	709.66	-	-	0.0
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	40.0	15.07	0.00	0.00	694.59	-	-	-
ADDITIONAL ΔV REQ'D	QUARANTINE ORBIT (PQ) - 2-BURN	220.0	22.8	0.0	0.0	0.0	0.0	7.30	0.00	0.00	687.29	4	885.0	-
RELAY Drag Dens 95%	OTM (Drag)	220.0	1.0	0.0	0.0	0.0	0.0	0.32	0.00	0.00	686.97	-	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	10.0	3.68	0.00	0.00	683.29	-	-	-
	SUB-TOTALS		59.7		1130.1		91.8	56.22	174.37	148.21				
	TOTAL MISSION ΔV		1281.6	S/C Dry Mass - Actual (kg)			673.28	S/C Dry Mass Capability (kg)			674.73	ΔDry Mass (kg)		1.45

Spacecraft Injected Mass Breakdown (kg)		Mission ΔV Breakdown		Prop Tanks	Load Error	Max Usable[1]	Desired Usable	Total Load Plus Error[2]	Full Tanks (100% Fill)[3]	Fill Level (Percent)	PROPELLANT (kg)	
S/C Dry Mass	673.28	Translational ΔV	1189.80								Usable	378.81
Dry Mass Margin	0.45	Rotational ΔV	91.80	N2O4	0.37	164.72	148.21	152.55	177.32	86.03%	Unusable 2%	7.58
Total Propellant	388.37	Bipropellant ΔV	1130.10	(2x) N2H4	0.62	229.29	230.59	235.82	246.94	95.50%	Load Error	0.99
Total Inj Mass	1062.10	Monopropellant ΔV	151.50			Ge (m/s^2)	9.80665	Engine Thrust (N)	BIPROP	658.7	Oxidizer Ballast	1.00
Uncertainties	+0.86 / -1.21	Desired Dry Mass Margin (kg)		0.45	BIPROP MIX RATIO		0.85		MPROP	4.45	Total Prop	388.37

[1]5% Ullage [2]Usable+Unusable+Ld_Err+Excess [3]Per Dominick Oxidizer Ballast (1=Y/2=N) 1

MSN DV - FMP BaseLoad

Figure D-2: ΔV *Budget, 6 November 1996 Launch*

MARS GLOBAL SURVEYOR SPACECRAFT MANEUVER EXECUTION
Mission ΔV Budget - ORR Baseline (8/96)
Launch Date - 11/6/96

SPACECRAFT INJECTED MASS (kg)	1062.10
Spacecraft Dry Mass (kg)	673.28
Dry Mass Margin (kg)	0.45
Total Propellant (kg)	388.37

Spacecraft Propellant Load

Engine	Thrust (N)	Propellant Tanks	Load Error	Max Usable*	Desired Usable	Usable Plus Error	Unusable 2%	Oxidizer Ballast	Tank Totals	Tank Redlines
BIPROP	658.7	N2O4 (kg)	0.37	164.72	148.21	148.58	2.96	1.00	152.54	4.33
MPROP	4.45	(2x) N2H4 (kg)	0.62	229.29	230.59	231.21	4.61	–	235.82	5.23

Ge (m/s²)	BIPROP MIX RATIO
9.80665	0.85

*5% Ullage

MISSION PHASE	MANEUVER	MONOPROP TRANSLATIONAL Isp (s)	MONOPROP TRANSLATIONAL ΔV Reqd (m/s)	BIPROP TRANSLATIONAL Isp (s)	BIPROP TRANSLATIONAL ΔV Reqd (m/s)	MONOPROP ROTATIONAL Isp (s)	MONOPROP ROTATIONAL ΔV Reqd (m/s)	PROPELLANT USAGE M-HYDRZ (kg)	PROPELLANT USAGE B-HYDRZ (kg)	PROPELLANT USAGE NTO (kg)	PROPELLANT REMAINING HYDRZ (kg)	PROPELLANT REMAINING NTO (kg)	POSTBURN S/C MASS (kg)	EST. BURN DURATION #T	EST. BURN DURATION (s)
CRUISE ΔV95	TCM-1/2	0.0	0.0	317.0	36.0	220.0	0.3	0.15	6.61	5.62	229.06	146.93	1049.72	n/a	57.7
	TCM-3/4	220.0	2.0	0.0	0.0	0.0	0.0	0.97	0.00	0.00	228.09	146.93	1048.75	4	117.9
	pre-Lnch ΔV Reserves	220.0	0.0	315.0	20.1	0.0	0.0	0.00	3.68	3.13	224.41	143.80	1041.95	n/a	27.0
CAPTURE	MOI (Rp = 3700 km, Per = 48 Hr)	0.0	0.0	317.0	968.2	220.0	5.9	2.85	150.72	128.12	70.85	15.69	760.26	n/a	1329.5
Aerobraking Walk-in	AB-1	0.0	0.0	315.0	7.5	220.0	0.1	0.04	1.00	0.85	69.81	14.84	758.38	n/a	8.6
	AB-i (i=2,4)	220.0	2.5	0.0	0.0	0.0	0.0	0.88	0.00	0.00	68.94	14.84	757.50	4	106.5
Aerobraking Main Phase	ABM Translation	220.0	5.0	0.0	0.0	0.0	0.0	1.75	0.00	0.00	67.18	14.84	755.75	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	5.0	2.03	0.00	0.00	65.16	14.84	753.72	-	-
Aerobraking Walk-out	ABM Translation	220.0	20.0	0.0	0.0	0.0	0.0	6.95	0.00	0.00	58.20	14.84	746.77	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	30.0	11.93	0.00	0.00	46.27	14.84	734.84	-	-
Aerobraking Contingency	Aerobraking Pop-Up	220.0	2.5	315.0	14.5	0.0	0.0	0.85	1.86	1.58	43.56	13.26	730.55	n/a	15.9
	post-Lnch ΔV Reserves	220.0	0.0	315.0	0.0	0.0	0.0	0.00	0.00	0.00	43.56	13.26	730.55	4	0.0
Transition To Mapping	ABX	220.0	0.0	317.0	60.2	220.0	0.4	0.14	7.57	6.44	35.85	6.82	716.40	n/a	66.3
	TMO	0.0	0.0	317.0	16.1	220.0	0.1	0.03	2.00	1.70	33.82	5.12	712.67	n/a	17.5
	OTM-1 (Frzn)	0.0	0.0	315.0	7.5	0.0	0.0	0.00	0.93	0.79	32.89	4.33	710.94	n/a	10.0
MAPPING Drag Dens 95%	OTM (Drag/GTE)	220.0	3.9	0.0	0.0	0.0	0.0	1.28	0.00	0.00	31.60	4.33	709.66	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	40.0	15.07	0.00	0.00	16.53	4.33	694.59	-	-
ADDITIONAL ΔV REQ'D	QUARANTINE ORBIT (PQ) - 2-BURN	220.0	22.8	0.0	0.0	0.0	0.0	7.30	0.00	0.00	9.23	4.33	687.28	4	885.0
RELAY Drag Dens 95%	OTM (Drag)	220.0	1.0	0.0	0.0	0.0	0.0	0.32	0.00	0.00	8.91	4.33	686.97	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	10.0	3.68	0.00	0.00	5.23	4.33	683.29	-	-
	SUB-TOTALS		59.7		1130.1		91.8	56.21	174.38	148.22	S/C Dry Mass (kg)		673.73		
	MISSION TOTAL ΔV		1281.6												

MSN DV - FMP BaseLoad

Figure D-3: Δ*V Budget, 25 November 1996 Launch*

<u>**MARS GLOBAL SURVEYOR SPACECRAFT MANEUVER EXECUTION**</u>
Mission ΔV Budget - ORR Baseline (8/96)
Launch Date - 11/25/96

SPACECRAFT INJECTED MASS (kg)	1062.10
Spacecraft Dry Mass (kg)	673.28
Dry Mass Margin (kg)	0.45
Total Propellant (kg)	388.37

Ge (m/s^2)
9.80665
BIPROP MIX RATIO
0.85

Spacecraft Propellant Load

Engine Thrust (N)		Propellant Tanks	Load Error	Max Usable*	Desired Usable	Usable Plus Error	Unusable 2%	Oxidizer Ballast	Tank Totals	Tank Redlines
BIPROP	658.7	N2O4 (kg)	0.37	164.72	148.21	148.58	2.96	1.00	152.54	4.33
MPROP	4.45	(2x) N2H4 (kg)	0.62	229.29	230.59	231.21	4.61	-	235.82	5.23

*5% Ullage

MISSION PHASE	MANEUVER	MONOPROP TRANSLATIONAL Isp (s)	MONOPROP TRANSLATIONAL ΔV Reqd (m/s)	BIPROP TRANSLATIONAL Isp (s)	BIPROP TRANSLATIONAL ΔV Reqd (m/s)	MONOPROP ROTATIONAL Isp (s)	MONOPROP ROTATIONAL ΔV Reqd (m/s)	PROPELLANT USAGE M-HYDRZ (kg)	PROPELLANT USAGE B-HYDRZ (kg)	PROPELLANT USAGE NTO (kg)	PROPELLANT REMAINING HYDRZ (kg)	PROPELLANT REMAINING NTO (kg)	POSTBURN S/C MASS (kg)	EST. BURN DURATION #T	EST. BURN DURATION (s)
CRUISE ΔV95	TCM-1/2	0.0	0.0	317.0	49.0	220.0	0.3	0.15	8.98	7.63	226.70	144.91	1045.34	n/a	78.4
	TCM-3/4	220.0	2.0	0.0	0.0	0.0	0.0	0.97	0.00	0.00	225.73	144.91	1044.37	4	117.4
	pre-Lnch ΔV Reserves	220.0	0.0	315.0	7.0	0.0	0.0	0.00	1.28	1.09	224.45	143.83	1042.01	n/a	9.4
CAPTURE	MOI (Rp = 3700 km, Per = 48 Hr)	0.0	0.0	317.0	975.1	220.0	5.9	2.85	151.65	128.90	69.96	14.93	758.61	n/a	1334.5
Aerobraking Walk-in	AB-1	0.0	0.0	315.0	7.5	220.0	0.1	0.04	0.99	0.85	68.93	14.08	756.74	n/a	8.6
	AB-i (i=2,4)	220.0	2.5	0.0	0.0	0.0	0.0	0.88	0.00	0.00	68.05	14.08	755.86	4	106.2
Aerobraking Main Phase	ABM Translation	220.0	5.0	0.0	0.0	0.0	0.0	1.75	0.00	0.00	66.30	14.08	754.11	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	5.0	2.02	0.00	0.00	64.28	14.08	752.09	-	-
Aerobraking Walk-out	ABM Translation	220.0	11.0	0.0	0.0	0.0	0.0	3.82	0.00	0.00	60.46	14.08	748.27	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	26.5	10.57	0.00	0.00	49.89	14.08	737.70	-	-
Aerobraking Contingency	Aerobraking Pop-Up	220.0	15.8	315.0	0.0	0.0	0.0	5.38	0.00	0.00	44.51	14.08	732.32	n/a	0.0
	post-Lnch ΔV Reserves	220.0	0.0	315.0	0.0	0.0	0.0	0.00	0.00	0.00	44.51	14.08	732.32	4	0.0
Transition To Mapping	ABX	220.0	0.0	317.0	65.8	220.0	0.4	0.14	8.29	7.05	36.08	7.03	716.84	n/a	72.6
	TMO	0.0	0.0	317.0	17.9	220.0	0.1	0.03	2.22	1.89	33.82	5.14	712.69	n/a	19.4
	OTM-1 (Frzn)	0.0	0.0	315.0	7.5	0.0	0.0	0.00	0.93	0.79	32.89	4.35	710.97	n/a	10.0
MAPPING Drag Dens 95%	OTM (Drag/GTE)	220.0	3.9	0.0	0.0	0.0	0.0	1.28	0.00	0.00	31.60	4.35	709.68	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	40.0	15.07	0.00	0.00	16.53	4.35	694.61	-	-
ADDITIONAL ΔV REQ'D	QUARANTINE ORBIT (PQ) - 2-BURN	220.0	22.8	0.0	0.0	0.0	0.0	7.30	0.00	0.00	9.23	4.35	687.31	4	885.0
RELAY Drag Dens 95%	OTM (Drag)	220.0	1.0	0.0	0.0	0.0	0.0	0.32	0.00	0.00	8.91	4.35	686.99	-	-
	ACS Rotation	0.0	0.0	0.0	0.0	190.0	10.0	3.68	0.00	0.00	5.23	4.35	683.31	-	-
	SUB-TOTALS		64.0		1129.8		88.3	56.24	174.35	148.20	S/C Dry Mass (kg)		673.73		
	MISSION TOTAL ΔV		1282.1												

MSN DV - FMP BaseLoad

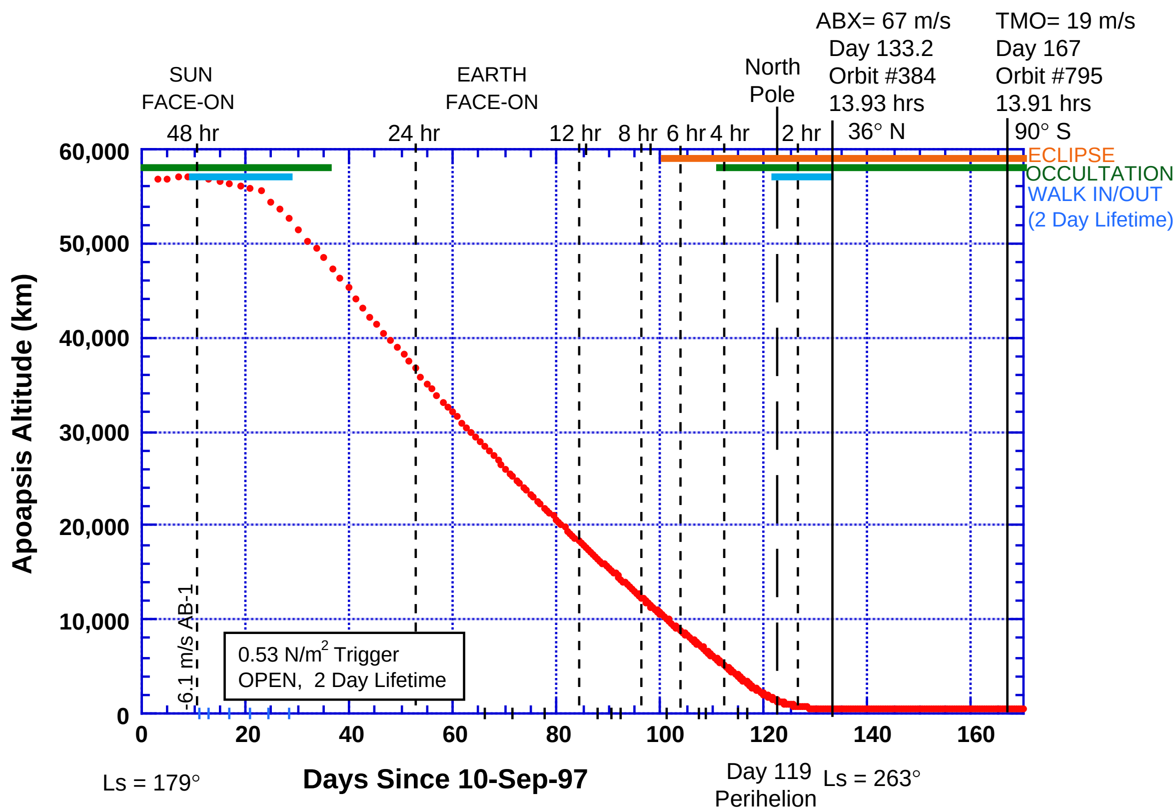
Appendix E - Aerobraking Design Data

This appendix provides detailed data regarding the reference trajectory design for aerobraking. The design presented in the mission plan assumes a launch on 6 November 1996, critical scale height approach to determine the size of the walk-in maneuvers (see Section 5), and a two-day orbit lifetime during aerobraking walk-out.

Table E-1: ***Trajectory Profile Summary for Aerobraking***

Event	Days From MOI	Date	Orbit Number	Periapsis / Apoapsis Alt. (km)
MOI	MOI+ 0	11-Sep-97	Orbit 1	300 / 56,675
Walk-in Maneuver AB1	MOI+ 9 days	20-Sep-97	Orbit 5	150 / 56,980
Walk-in Maneuver AB2	MOI+ 11 days	22-Sep-97	Orbit 6	133 / 57,000
Walk-in Maneuver AB3	MOI+ 15 days	26-Sep-97	Orbit 8	124 / 56,640
Walk-in Maneuver AB4	MOI+ 19 days	29-Sep-97	Orbit 10	118 / 56,040
Walk-in Maneuver AB5	MOI+ 23 days	3-Oct-97	Orbit 12	113 / 55,510
Walk-in Maneuver AB6	MOI+ 27 days	7-Oct-97	Orbit 14	112 / 53,600
Start of Main Phase	MOI+ 28 days	8-Oct-97	Orbit 15	112 / 52,650
Orbit Period at 40 Hours	MOI+ 32 days	13-Oct-97	Orbit 18	112 / 49,460
Orbit Period at 30 Hours	MOI+ 46 days	27-Oct-97	Orbit 28	113 / 39,820
Orbit Period at 24 Hours	MOI+ 57 days	7-Nov-97	Orbit 37	113 / 33,170
Orbit Period at 12 Hours	MOI+ 83 days	3-Dec-97	Orbit 74	114 / 18,610
Orbit Period at 6 Hours	MOI+ 102 days	22-Dec-97	Orbit 129	112 / 8,980
Orbit Period at 3 Hours	MOI+ 115 days	4-Jan-98	Orbit 204	104 / 3,190
Begin Walk-Out	MOI+ 121 days	10-Jan-98	Orbit 256	108 / 1,390
Aerobrake Termination Burn	MOI+ 132 days	21-Jan-98	Orbit 385	426 / 450

Figure E-1: ***Apoapsis Altitude vs. Time (top), Local Mean Solar Hour vs. Time (bottom)***



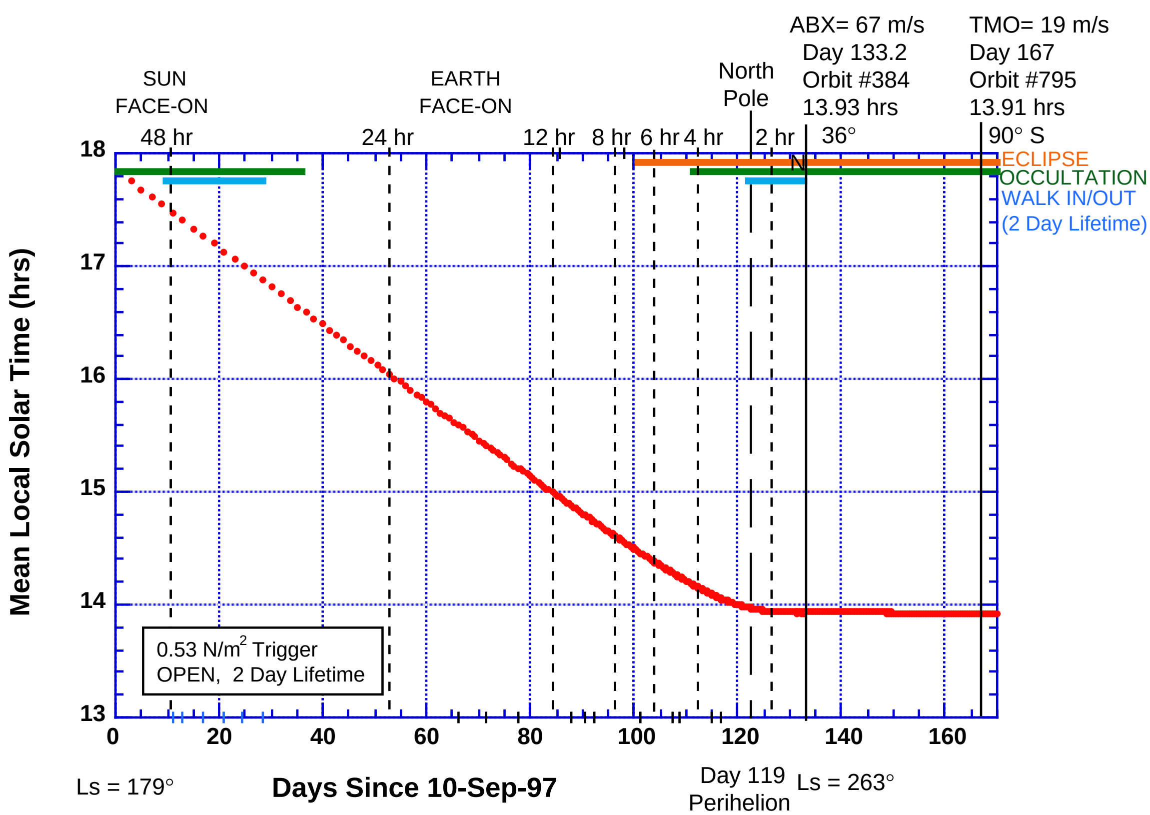
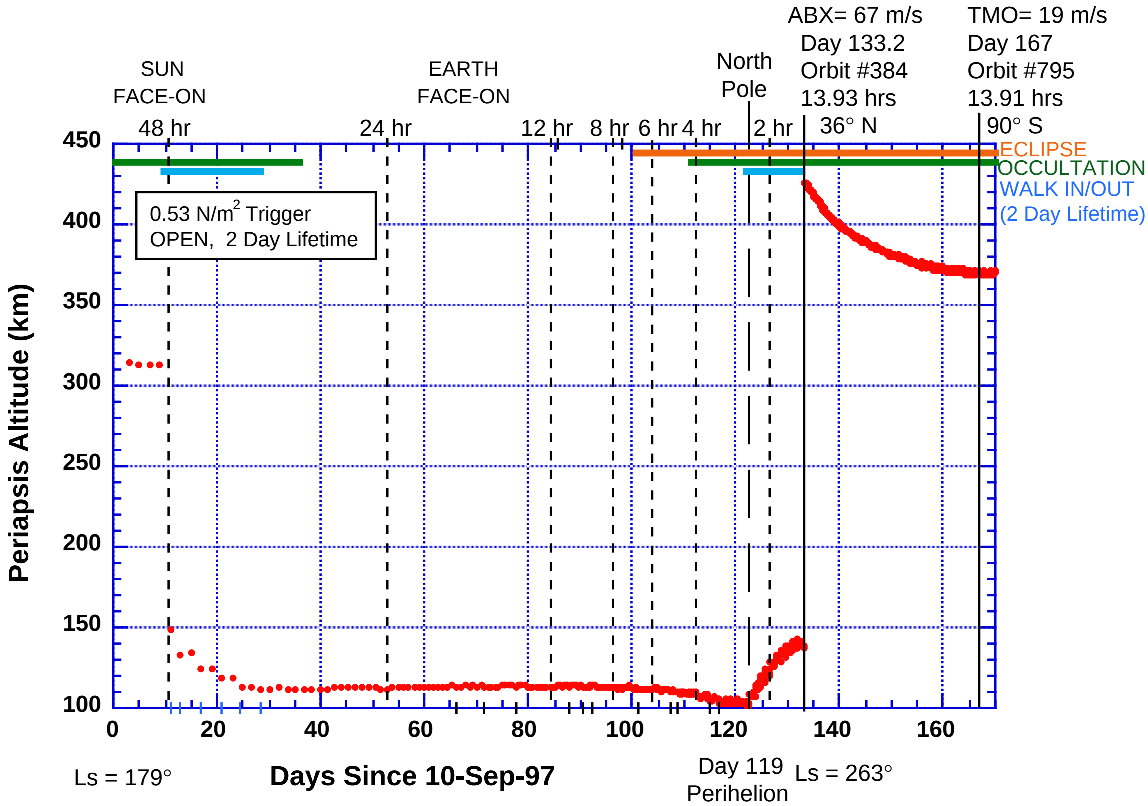


Figure E-2: ***Periapsis Altitude vs. Time (top), IAU Argument of Periapsis vs. Time (bottom)***



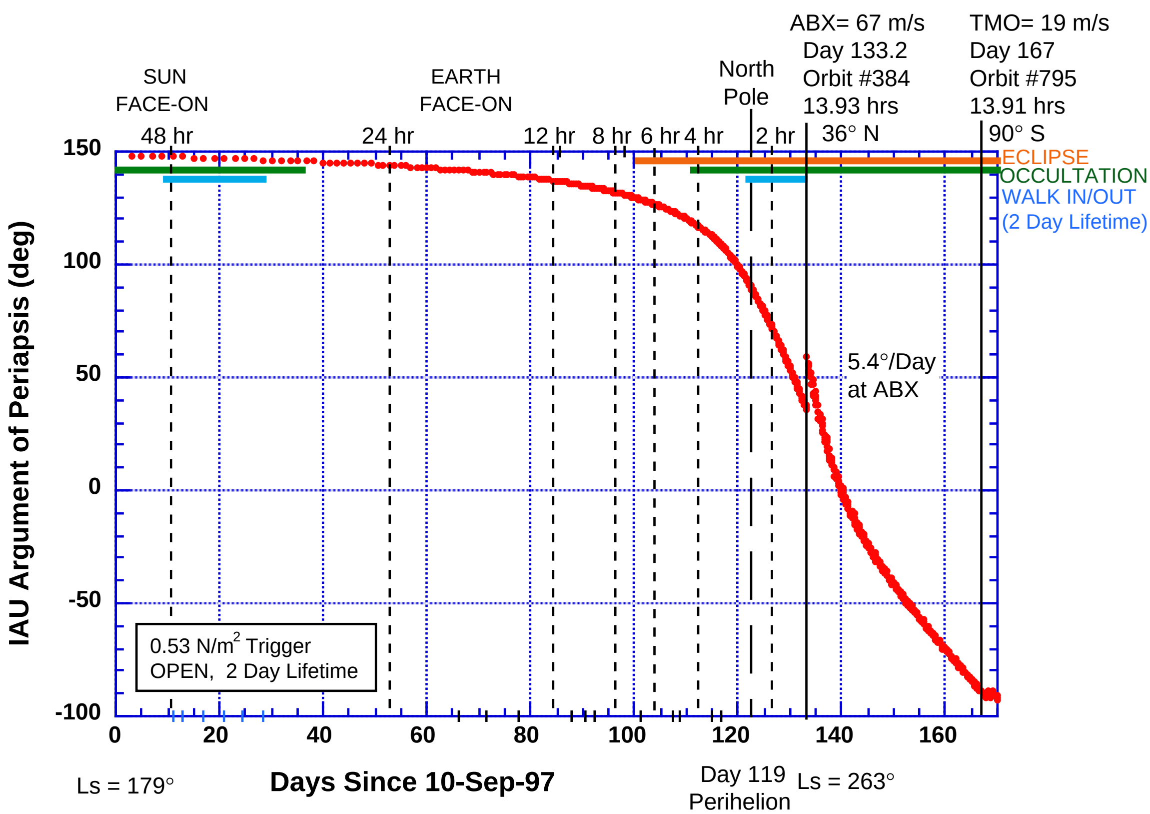


Figure E-3: ***Heating Rate at Periaspis (top), Entry / Exit Times (bottom)***